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SEPTEMBER 29, 1969

(NASA-CR-158403) SPACE SHUTTLE DATA:  
TWO-STAGE VEHICLE SUMMARY (Lockheed Missiles  
and Space Co.) 66 p

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## SPACE SHUTTLE DATA

### TWO-STAGE VEHICLE SUMMARY

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LMSC-A959125  
September 29, 1969

SPACE SHUTTLE DATA  
TWO-STAGE VEHICLE SUMMARY

## FOREWORD

The purpose of this document is to present pertinent details on LMSC's current Space Shuttle design and to summarize the nine-volume data package transmitted to NASA on September 12, 1969 (LMSC-A955317A).

Design details and systems cost are presented in Section 1; and the data summaries are contained in Section 2, in which subsections correspond to volumes of the earlier submittal. The subsection on structures (2.3) has been updated to be compatible with the design presented in Section 1.

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## Section 1 CURRENT DESIGN

LMSC's primary design objective is to lay the groundwork for a cost-effective Space Shuttle that will not only fulfill basic mission requirements but also afford NASA the option of extending mission application without significant cost penalties. This inherent mission flexibility results from the increased maneuverability of the vehicle and its potential cross-range capabilities.

The LMSC 50,000-lb payload Two-Stage Space Shuttle consists of two vehicles operated independently. The booster engines operate at 100 percent rated thrust during ascent, while the orbiter engines operate at 10 percent of nominal flow rate during the first-stage operating period. Although this method of operation incurs a performance penalty, it assures orbiter engine ignition, which contributes specifically to personnel safety.

The baseline launch vehicle is sized for the basic NASA mission, as indicated by the following requirements:

● Payload weight	50,000 lb up/down
● Payload size	15 ft dia, 60 ft length/22 ft dia, 30 ft length, composite*
● Ascent orbit	Perigee - 45 nm Apogee - 100 nm Inclination - 55 deg
● Post insertion $\Delta V$	2,000 ft/sec**
● Crew	2 men
● Mission duration	7 days
● Return cross range	400 nm

\*The payload dimensions are assumed to be either 15 by 60 or 22 by 30, and the unused cargo compartment volume in either case is assumed to be usable for propellant tankage.

\*\*Of this, 142 fps is derived from the orbital RCS, while the remainder comes from primary propulsion.

- |                     |   |
|---------------------|---|
| ● Launch conditions | Booster engines at 100 percent thrust           |
|                     | Orbiter engines at 10 percent flow rate         |
| ● Engine thrust     | 400,000 SL or 600,000 SL each                   |
| ● Landing approach  | 150 to 180 knots with go-around for both stages |

### Launch Vehicle Configuration

The LMSC Two-Stage launch vehicle configuration is presented in Drawing SKG092569.\* As shown, the launch configuration has a delta lifting body mounted on the upper side of the delta winged booster. The vehicle is configured for a vertical takeoff with all engines ignited at liftoff and with horizontal landing of the individual stages. Touch-down velocities are consistent with those of high-performance aircraft, and the capability to pull up and go around the target airfield is incorporated into each stage.

### Booster

The general arrangement of the booster is shown in Drawing SKG092669. As shown, the booster comprises a ring-stringer stiffened cylindrical fuselage with a multispar and rib delta wing. The oxidizer and fuel tanks, which represent the major portion of the fuselage, are designed as the primary load-carrying structure to minimize weight. Presently, no propellant is required to be stored within the wing structure. The wing is attached to the fuselage according to a structural arrangement that permits the fuselage to expand and contract longitudinally without constriction by the wing. The booster has multiple rocket engines (14 at 400,000-lb SL thrust), mounted in the aft end and gimbaled for ascent control. The expansion ratio is 35:1. The four cruise-back jet engines, which generate a total of 115,400 pounds of sea level thrust, are pod-mounted on the upper aft fuselage.

### Orbiter

The orbiter (shown in Drawing SKG092769), is a delta lifting body with three rocket engines, mounted at the aft end of the fuselage. These engines are similar to the booster engines, but they have two-position nozzles at 35:1 to accommodate atmospheric

\*Drawings are placed at the end of this section.

operation and 150:1 for high-altitude operation. The four jet engines, which deliver a total of 73,250 pounds of operating thrust, are located in the upper aft fuselage in the concave section between the fins and the body. They are extended when required for subsonic flight. (An optional installation is also shown.) The orbiter requires thrust augmentation and propellant kits and refueling to satisfy ferry requirements.

### Propulsion/Performance Characteristics

Pertinent propulsion and performance data are presented in the following tables. The propulsion data are for high pressure (3000 psi) engines with bell nozzles; the first stage nozzle is fixed, and the orbiter has a two-position nozzle. All engines have the same turbine machinery.

#### PROPULSION/PERFORMANCE CHARACTERISTICS

	<u>Booster</u>	<u>Orbiter</u>
Propellant/Mixture Ratio	← 7:1 →	
Expansion Ratio (sea-level vacuum)	35:1 SL	35:1 SL & 150:1 vac
Specific Impulse (SL/vac)	387.8 SL/428.5 vac	378.8 SL & 454.2 vac
Propellant Fraction	0.850	0.766
Ideal $\Delta V$ (fps)	14,063	18,312
Mass Ratio	2.78	3.39
Number of Engines	14	3
Thrust - Sea Level	5430K	
- Vacuum	6007K	1332K
Flyback Engine System		
Number/Type	4 turbofan	4 turbofan
Operating Thrust	115,400	73,250
SFC (lbm/lbf-hr)	0.43	0.50
Fuel (lb)	38,130	3660

# VEHICLE VELOCITY REQUIREMENTS

*Ascent Ideal $\left(\frac{45}{100}\right)$ nm , 55°	Total (Subtotal)	Booster	Orbiter
	30,150	13,667	16,483
<b>Losses</b>			
Drag	-936	-915	-21
Gravity (inc's pot. energy)	-3761	-3580	-181
Back Pressure	-278	-252	-26
Thrust Alignment Maneuver }	-158	-9	-149
		4756	
Earth Rotation (gain) (equiv initial vel)	+871	+871	-
Flight Performance res (3/4% ΔV)	226		226
<b>Phasing</b>			
Transfer 100 to 270 nm		13,667	
Circularize		+750	
Rendezvous } 142 ft/sec			
Docking } at ≤ 1 ft/sec <sup>2</sup>	2,000	871	2,000
Orbit Maneuvers		9,100	
Undocking			
Deorbit			
Orbit Contingency			
<b>Total Ideal Velocity</b>	<b>32,376</b>	<b>13,667</b>	<b>18,709</b>

\*Based on vacuum performance, this includes orbit velocity + potential energy gained + losses - earth rotation velocity vector.

Weight Summaries

The weight summary is presented in the following table. Launch weight is about 3.74 million pounds if engine thrust is limited to 600,000 pounds and 3.81 million pounds for a 400,000 pound thrust limit. The booster and orbiter landing weights are 373,000 and 259,000 pounds, respectively.

<u>Function or Condition</u>	<u>Booster</u>	<u>Orbiter</u>	<u>Stage</u>
Aerodynamic Surfaces			
Wing	57,983	—	
Fin	{ Included in Wing }	4,599	
Elevon		8,996	
Body Structure			
Shell and frames	67,467	44,212	
Thrust structure	13,141	2,704	
Pressurized compartment	1,350	1,350	
Induced Environment Protection			
Body	19,039	30,606	
Wing	8,166	—	
Fin	—	4,654	
Launch, Recovery, and Docking			
Launch gear	15,000	5,000	
Landing gear	14,962	10,316	
Docking structure	—	500	
Main Propulsion			
Liquid rocket engine (<600K)	57,715	11,940	
Air-breathing engine	32,522	20,526	
Rocket fuel container	{ Integral }	13,372	
Rocket oxidizer container		10,635	
Rocket propellant systems	11,735	2,321	
Jet fuel container	2,884	277	
Jet propellant systems	455	433	

<u>Function or Condition</u>	<u>Booster</u>	<u>Orbiter</u>	<u>Stage</u>
Orientation Controls, Sep., and Ullage	4,410	2,040	
Prime Power Source	447	1,026	
Power Conversion and Distribution	8,578	4,072	
Guidance and Navigation	842	842	
Instrumentation	200	200	
Communication	141	141	
Environmental Control	1,180	1,227	
Personnel Provisions	636	636	
Crew Station Controls and Panels	612	612	
Range Safety and Abort	200	200	
Contingency	31,966	18,344	
Dry Weight	351,631	201,781	553,412
Personnel	576	576	
Cargo	—	50,000	
Residual/reserve prop. and reactants	20,850	6,542	
ECS and prime power reactants	427	1,309	
Reaction control propellants	2,000	6,273	
Jet engine propellants	38,131	3,662	
Rocket engine propellants	2,348,058	704,916	
Launch Weight	2,761,673	975,059	3,736,732*
Less main propulsion propellants	-2,348,058	-46,612	-2,394,670
Stage Burnout Weight	413,615	928,447	1,342,062
Less main propulsion propellants	—	-619,658	
Booster/Orbiter Burnout Weight		308,789	
Less: ECS and prime power reactants	-127	-484	
Reaction control propellants	-2,000	-6,273	
Orbit and retro propellants	—	-36,646	
Entry Weight	411,488	263,386	
Less: Prime power reactants	-300	-825	
Jet engine propellants	-38,131	-3,662	
Landing Weight	373,057	258,899	

\*Using 400,000-pound thrust engines in lieu of 600,000 pound engines will increase launch weight to 3,812,582 pounds.

### Cost Considerations

Parametric cost estimates were made for the 3.7 million pound configuration. Costs were generated by using CERs defined in Annex 6 of the Air Force Space Transportations System Study, dated August 7, 1969. Operations costs are based on 100 flights per year over an operational period of 10 years.

The following assumptions are applicable:

<u>AFST Element No.</u>	<u>Element</u>	<u>Parameter</u>	<u>Quantity</u>
20	Ground test hardware	Number of equivalent units	4
21	Flight test hardware	Number of equivalent units	3
24a	Vertical flight test	Number of vertical flights	25
24b	Horizontal flight test	Number of horizontal flights	40
27	Ground equipment	Number of sets	1
32a	Flight crew training	Number of men	40
32b	Ground support training	Number of men	200

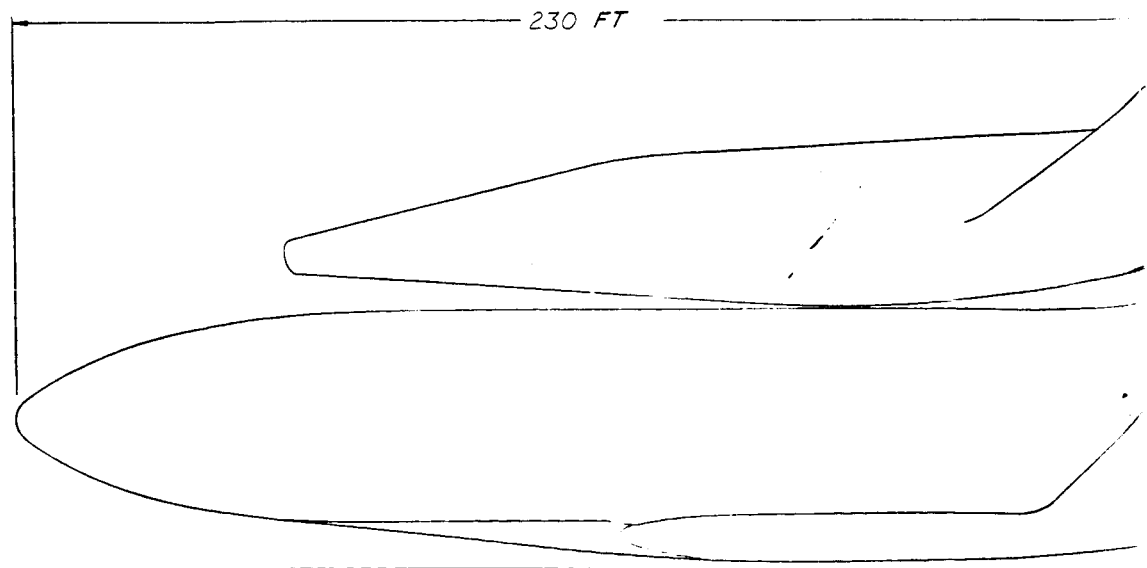
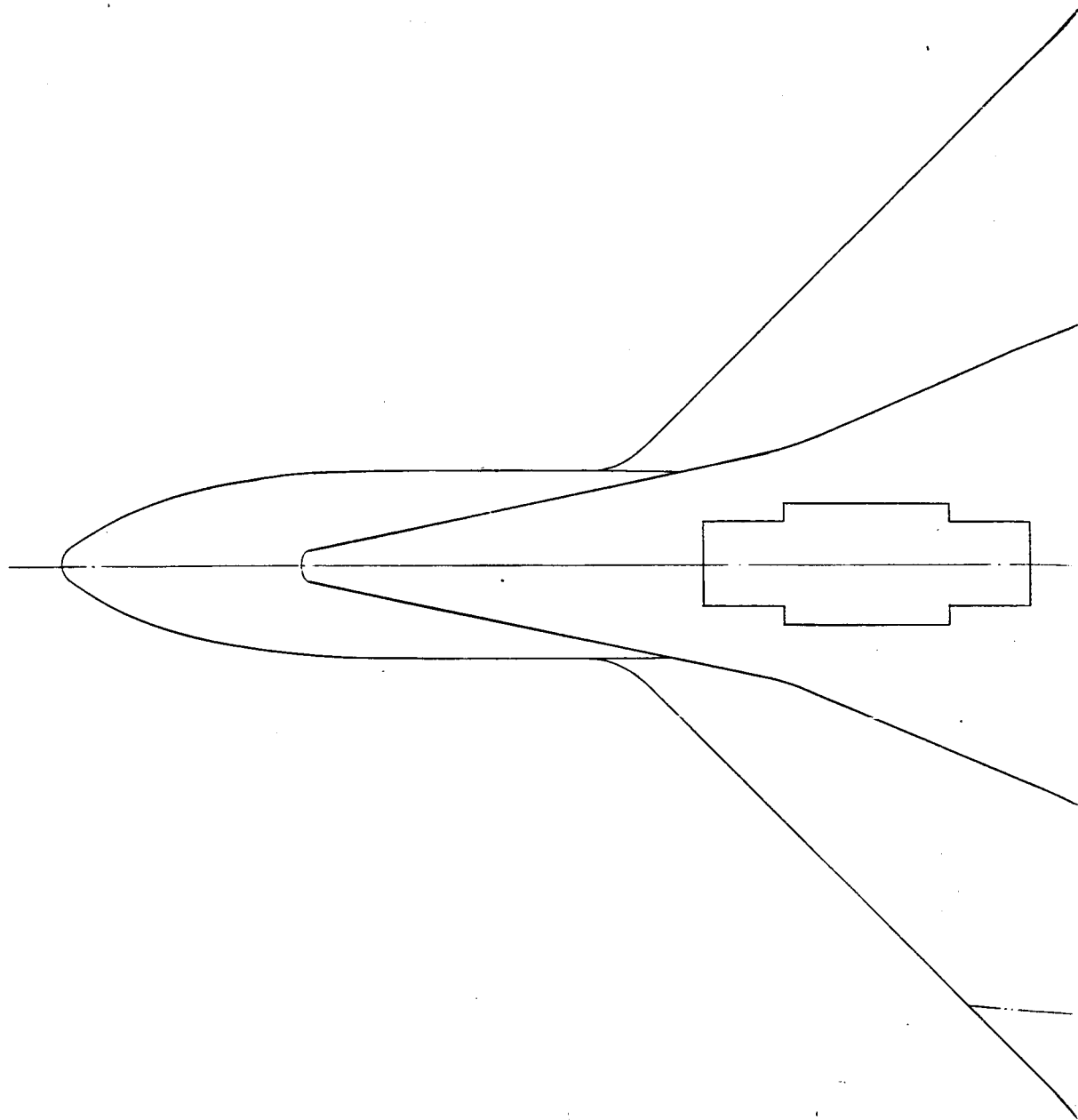
Further assumptions are that both orbiter and booster will use existing air-breathing engines for flyback. Therefore, only modification costs are included in Cost Element Number 9. In determining development costs for the booster subsystems, it is assumed that commonality percentages exists between the orbiter and the booster as follows:

Number 6	thermal protection	80
Number 12	reaction control	80
Number 13	electrical power	50
Number 14	avionics	80
Number 15	environmental control	80

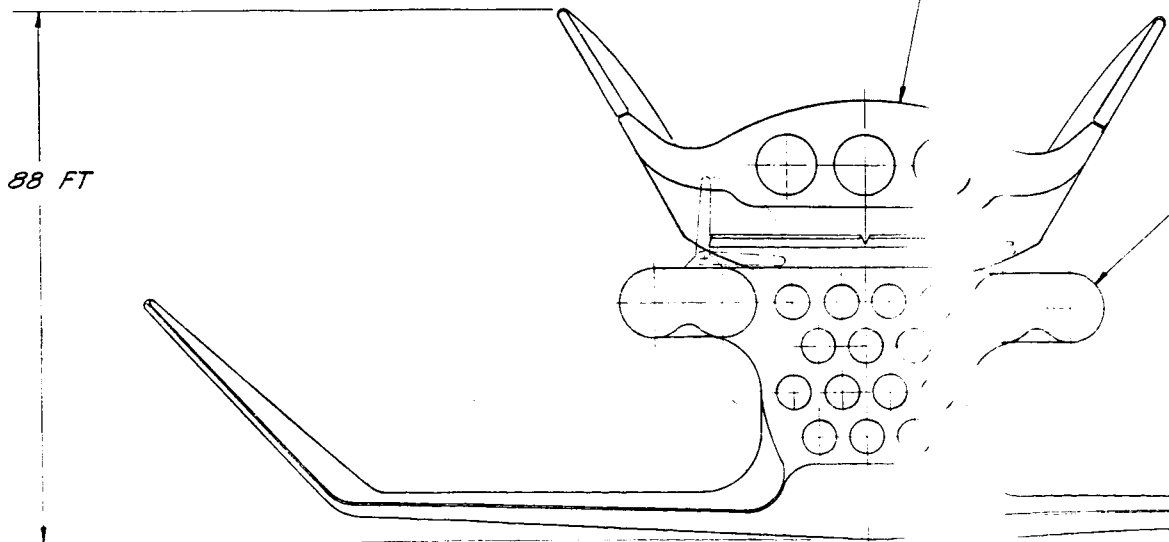
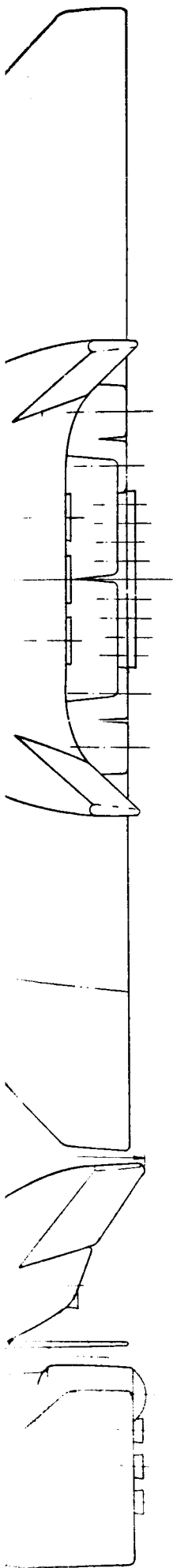
First unit cost for airframe structure (cost element number 75) is based on aluminum skin/stringer/ring construction with a complexity factor of 1.0.

The resulting cost estimates in millions of dollars are summarized as follows:

Subsystem development	1184.0
AGE development	59.2
Test hardware	1702.4
Test operations	520.0
Tooling and facilities	313.8
Training and simulators	20.5
Integration, G&A and fee	<u>1420.3</u>
Total RDT&E	<u>5220.2</u>
First unit flight system	243.2
Total operations costs (10 years)	2877.0
Average operations cost/flight (at 100 flights/year)	2.88



230 FT



SPACECRAFT  
REF: DWG SKS092769

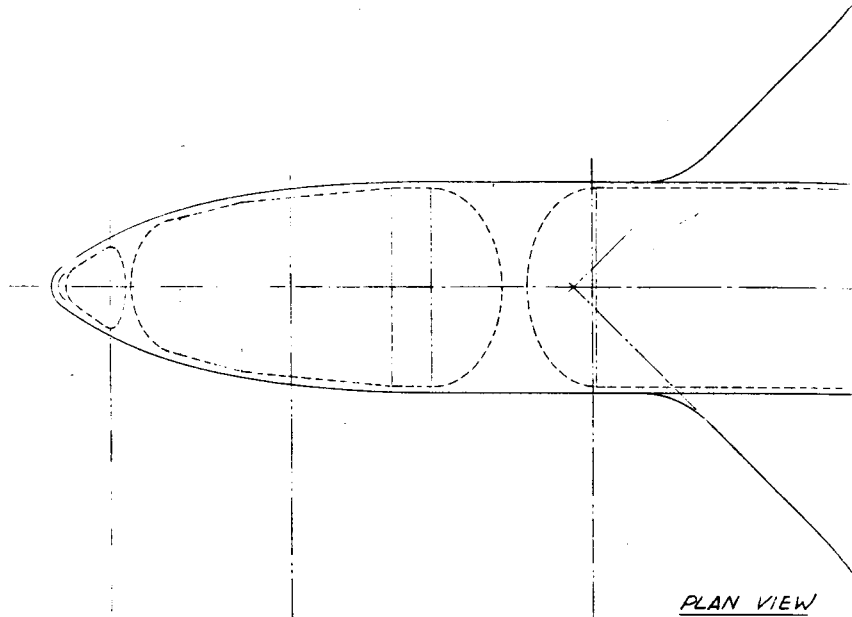
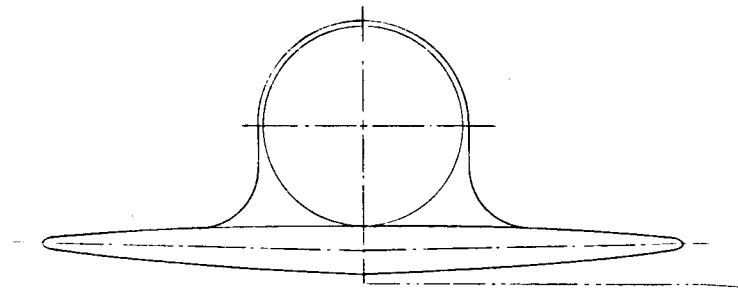
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SCALE, FT

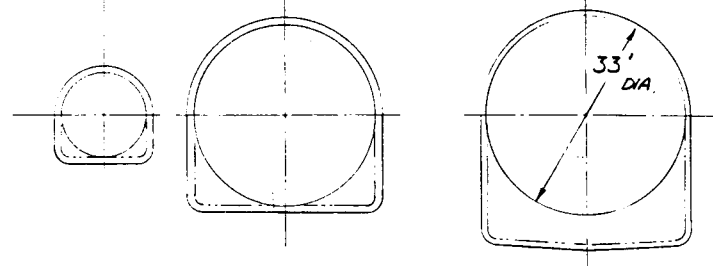
BOOSTER

REF: DWG SKG092669

DATE	4-27-60	LOCKHEED MISSILES & SPACE COMPANY	
DR	77	A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA	
APPD		LAUNCH VEHICLE -	
APPD		2-STAGE CARTER	
ENGRG		50 K PAYLOAD	
CHK			
APPD		SIZE CODE IDENT	DRAWING NO. SKG092562
APPD			REV

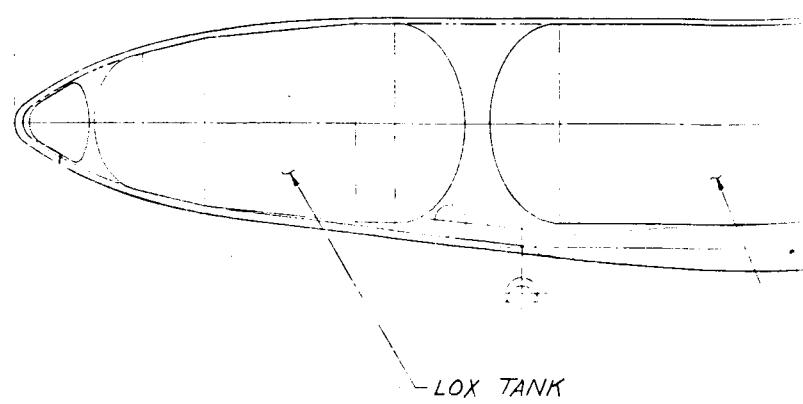


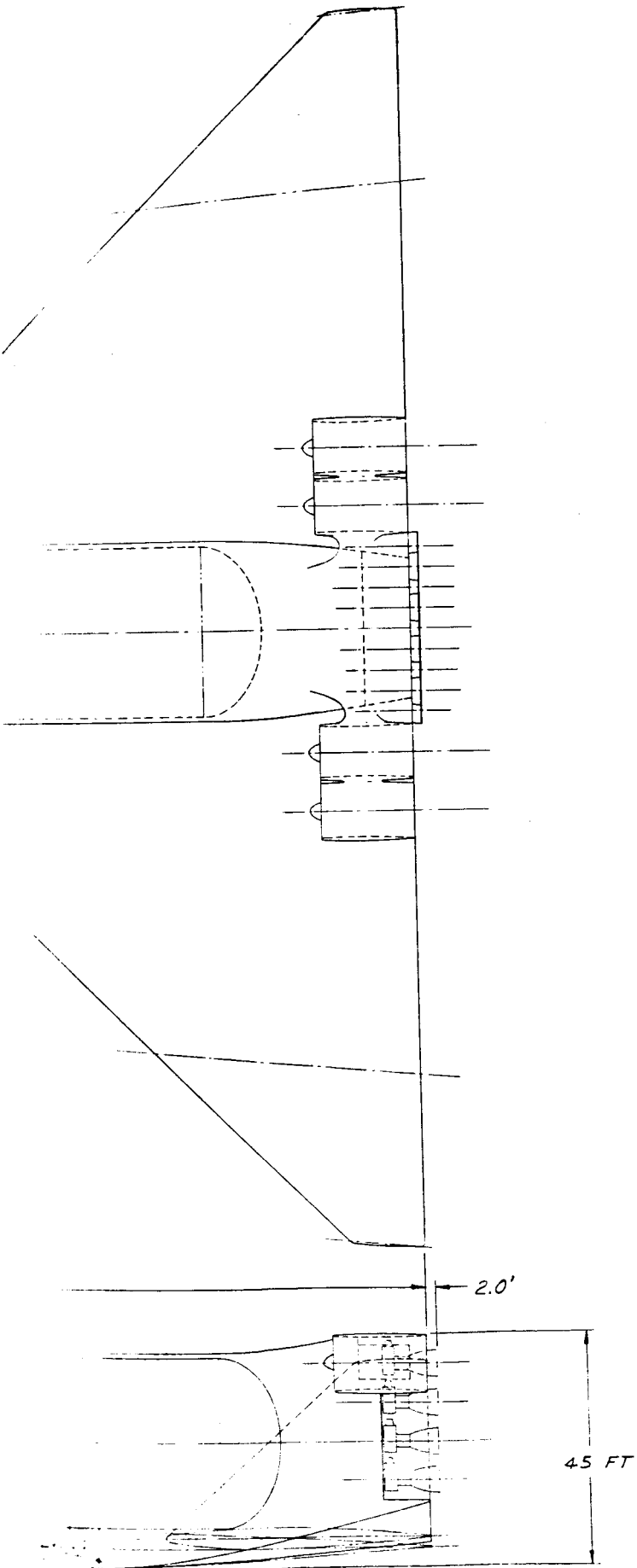
PLAN VIEW



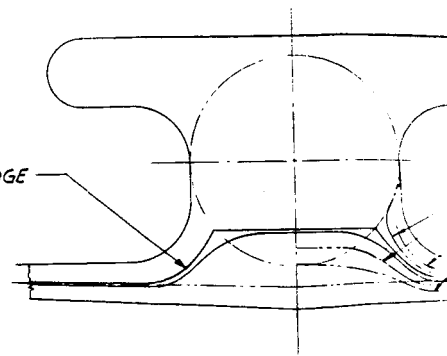
STA 0

228 FT

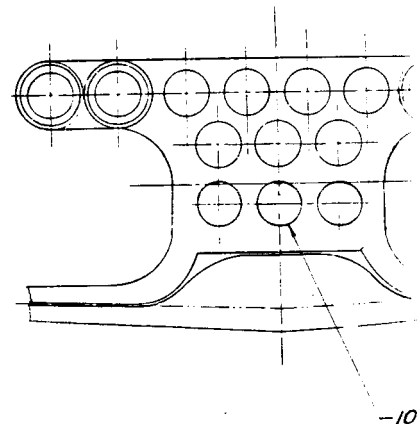




TRAILING EDGE  
STA 220



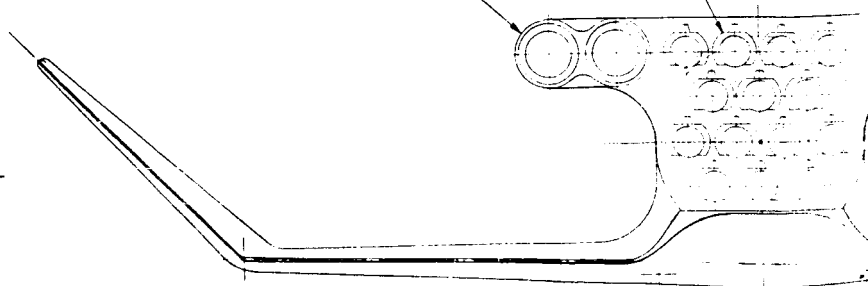
AFTERBODY CONTOUR



14 400 K SEA LEVEL  
THRUST ENGINES

ALTERNATE

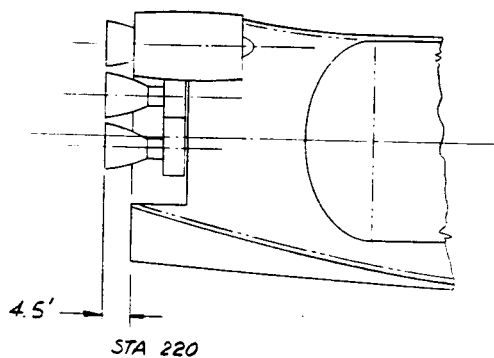
4 FANJET ENGINES



REAR VIEW

STA 210

STA 200



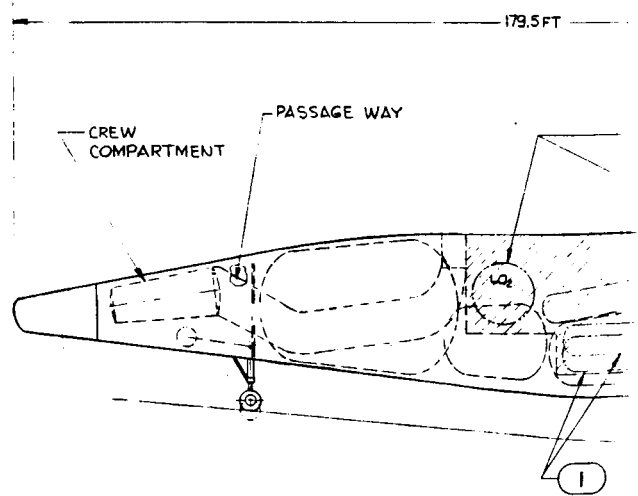
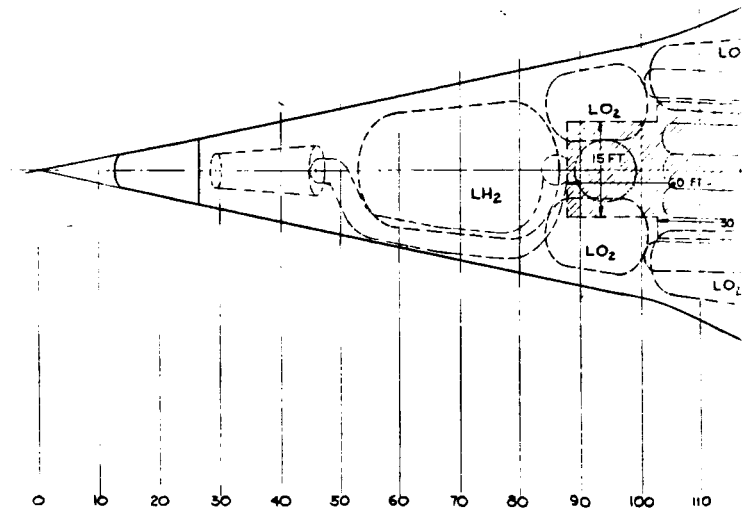
4.5'

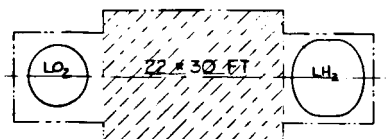
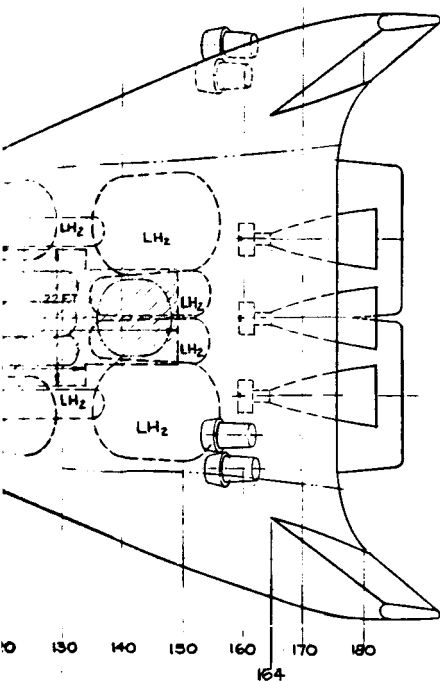
STA 220

SEA LEVEL  
EYES

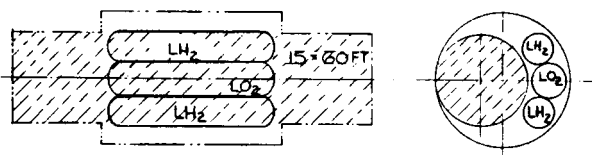
INSTALLATION

DATE	9-26-69	LOCKHEED MISSILES & SPACE COMPANY		
DR	775	P. O. BOX 100 DIVISION OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA		
APPRO		BOOSTER, 2-STAGE SYSTEM		
APPRO		50 K PAYLOAD		
APPRO		SIZE	CODE IDENT	DRAWING NO
APPRO				516092669
APPRO		SCALE	7:120	SHEET

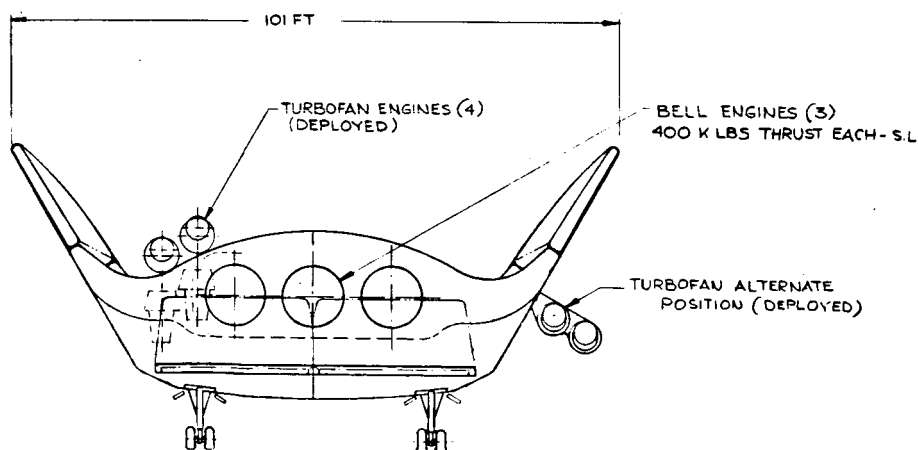
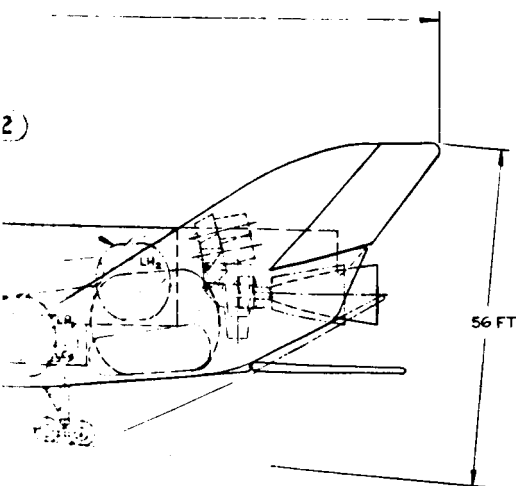




② TANK ARRANGEMENT FOR 22 FT DIA × 30 FT LG. P/L



① TANK ARRANGEMENT FOR 15 FT DIA × 60 FT LG. P/L



DATE	SEPT. 22, 1969	LOCKHEED MISSILES & SPACE COMPANY		
DR	tz Schmidt	A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION		
		SUNNYVALE, CALIFORNIA		
APPD		SPACECRAFT		
APPD		2 STAGE SYSTEM		
ENGRG		GENERAL ARRANGEMENT		
CHK				
APPD		SIZE	CODE IDENT	DRAWING NO.
APPD		E		SKS 092769
		SCALE 1/120		SHEET 1 OF 1

### NOTICE

The data contained in Section 2 of this report have been extracted from LMSC-A955317A, Vols. I - IX and, with the exception of Subsection 2.3, represent information that has not been updated. Hence, a correlation does not necessarily exist between the data of Sections 1 and 2.

The subsection on structures (2.3), however, has been updated to provide current information on the structural concepts used in the latest design reported in Section 1.

## Section 2

### CONCEPT DATA SUMMARY

#### 2.1 DESIGN INTEGRATION

The fully reusable Two-Stage shuttle concept summarized in this report consists of a modified delta lifting body orbiter, launched by a fixed-wing booster in a modified parallel-burn configuration. Major system characteristics are presented in the following table. These data relate to a system capable of carrying 50,000 pounds of payload to and from a 270-nm, 55-deg inclination orbit, with a 2000-ft/sec capability beyond the 45 by 100 nm reference orbit.

Table 2-1

#### MAJOR SYSTEM CHARACTERISTICS

Characteristics	Booster	Orbiter
Cargo Size	—	15' x 60'
Crew Size	2	2
Acceleration Limit	4g (crew); 3g passengers	
Cross Range	—	1630 at 2200°F
Hypersonic L/D	0.53 at 60°	2.0
Subsonic L/D	6.5	4.66
Overall Length	227 ft	162 ft
Liftoff Weight	3,033,417	1,031,711
Dry Weight	409,356	210,276
Propulsion System		
Mixture ratio	7:1	7:1
Specific impulse	392 SL/428.5 VAC	388.4 SL/454.2 SL
Thrust level	5880 KSL/6500K VAC	1180K
Number of engines	11	2
Ideal $\Delta V$	13,667 ft/sec	18,707 ft/sec

### Vehicle Sizing

Vehicle sizing has been accomplished by the use of a versatile vehicle synthesis program called MAGIC. This program is capable of simulating any geometrical configuration by adjusting appropriate scaling constants. All assumptions made in the program are continually checked and updated by configuration design, structures, weights, propulsion, thermodynamics, and performance groups.

In MAGIC, both the orbiter and the booster are grown photographically in the variation of size; i. e., all dimensions maintain the same ratio to length as the vehicle size changes, except for certain fixed volumes, such as crew cabin, payload bay, and a portion of the main engine compartment. MAGIC gives complete definition of such systems as reaction control, electrical power, environmental control, and landing improvement (jet engines or variable-geometry wings). It is capable of handling various throttling modes for the main engines, including differential throttling, parallel burning with crossfeed, parallel burning without crossfeed, and tandem (sequential) burning. The performance equations in MAGIC are backed up by extensive trajectory analysis over a wide range of parameters. MAGIC has proved to be a reliable, rapid way of evaluating a multiplicity of configurations.

### Parametric Sensitivity Studies

A study of the sensitivities of important characteristics covered:

- Staging velocity
- Launch thrust/weight
- Oxidizer/fuel ratio, orbiter and booster
- Ascent payload
- Return payload
- Total mission ideal velocity
- Propellant packaging, orbiter and booster
- Contingency
- Weight/area on orbiter

Staging-Velocity. Effect on system launch weight is shown in Fig. 2-1. Staging velocity is varied by varying the relative size of boost and orbital elements.

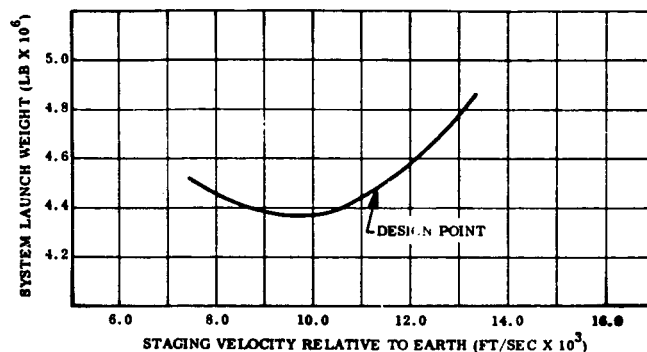


Fig. 2-1 System Launch Weight vs Staging Velocity

Launch Thrust/Weight. System launch weight tends to minimize at a  $(T/W)_0$  value around 1.55. A design point of 1.45 was selected to allow for growth capability by increasing thrust with minimum system effects.

Propellant Mixture Ratio. Oxygen/hydrogen mixture ratios from 6 to 8 were examined. Although system launch weight was found to be relatively insensitive to mixture ratio, it tended to minimize for orbiter and booster mixture ratios of 7 and 6.5, respectively.

Ascent and Return Payload. Although ascent payload is a major factor determining launch weight, it has little effect on wing loading; return payload has little effect on launch weight. All payloads were placed in a 15 ft dia, 60 ft long bay.

Mission Velocity. For the vehicle sizes and velocities considered, 1 ft/sec increases launch weight by about 500 pounds. Dry weights increase by around 40 pounds and 115 pounds for booster and orbiter, respectively.

Contingency. A given percentage of structural weight is to be added as a margin to account for possible vehicle growth. Sensitivity of system launch weight for both concepts was found to be about 75,000 lb/percent contingency.

Weight/Unit Area. Uniformly increasing weight of all wetted surface of the orbiter increases system launch weight by approximately 500,000 lb/lb/ft<sup>2</sup>.

**Effect of Cross Range on Sizing.** Three extreme cases were considered on the basis of whether all cross range is obtained by hypersonic glide, orbital plane change, or by subsonic cruise. A lifting body having hypersonic L/D of 2 and a wing-body having hypersonic L/D of 0.5 to 1.0 were investigated. Figure 2-2 shows launch weight versus cross range and indicates that hypersonic cross range is always cheaper than range obtained by subsonic cruise or by orbit plane change.

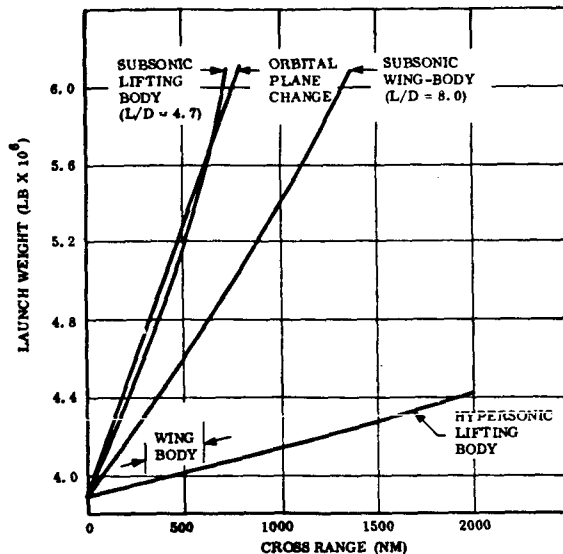


Fig. 2-2 System Launch Weight vs Orbiter Cross-Range Requirements

### Crew System Studies

The continuing effort in this area is specifically directed toward analysis and evaluation of the various candidate vehicle configurations.

Studies and analyses conducted to date have resulted in the tentative and preliminary identification of the following important conclusions:

- An environmental control/life support system is within the state-of-the-art, and a two-gas shirtsleeve environment is feasible. System weight and powers are respectively 1160 lb and 1360 peak watts for the orbiter and 430 lb and 610 peak watts for the booster.
- A modular environmental control system appears to be desirable for the ferry mode; a simple plugin, pullout concept should be used.
- Flight-crew face masks with an oxygen source should be provided at flight station for fire and smoke emergencies, together with an inert gas purge.

- The volume available for the flight-crew compartment is more than adequate.
- The orbiter vehicle flight-crew compartment should have seat- or capsule-eject capability in the R&D vehicle.
- The boost vehicle crew compartment should have seat- or capsule-eject capability for development and operational phases.
- The orbiter vehicle atmosphere should be able to increase from the design value of 10 psi total of 14.7 psi to be compatible with the space station environment.
- The crew flight-station console should be similar in configuration for the orbiter and the booster.
- The flight station for the two-man crew should have similar left/right-display/control capability with a central, shared panel.
- A docking station is required if the pilot or copilot cannot view directly through their normal out-of-vehicle visual system.
- A hygiene and food/water management station is extremely desirable for a flight compartment of the vehicle when in a 30-day orbit.
- A comfortable crew-compartment environment (e. g. , adequate exercise and movement volume) is highly desirable and feasible.
- Crew eject is feasible only at a low altitude within the atmosphere, subsonic speeds or on the pad.

## 2.2 AERODYNAMICS

### Aerodynamic Characteristics

Aerodynamic characteristics (subsonic through hypersonic) for the orbiter are based on wind tunnel test data and computerized arbitrary body analyses.

As shown in Fig. 2-3 (a) and (b), subsonic trim characteristics indicate that the configuration is longitudinally and directionally stable. Longitudinal stability margins of approximately 5 to 8 percent of the vehicle length are available, depending on attitude, as indicated in Fig. 2-3(a). Experimental data given in Fig. 2-3(c) show that trim angles of attack up to 23 deg are achievable for normal force coefficients ( $C_N$ ) to 0.6. Trimmed  $(L/D)_{\max}$  is 4.7 at 17-deg angle of attack, as shown in Fig. 2-3(d). Recent subsonic test data indicate that  $L/D$ s exceeding 5.0 are attainable with additional development.

Supersonic and hypersonic estimates have been based primarily on computer analyses. However, experimental data recently acquired at MSFC indicate a longitudinally and directionally stable configuration with sufficient trim authority to enable the vehicle to trim at  $L/D_{\max}$ . Test results show that maximum trimmed  $L/D$  is approximately 2.0 in the transonic-supersonic speed regime. Estimates from the hypersonic arbitrary body computer program indicate a stable-trimmed configuration from 10 to 70 degrees angle of attack during entry, as shown in Fig. 2-3(a). The trimmed spacecraft longitudinal stability margin is 4 percent of the vehicle length with a maximum  $L/D$  of 2.05, as indicated in Fig. 2-3(a) and (d). At a 60-deg entry attitude,  $L/D$  is 0.57. The entry spacecraft is directionally stable at angles of attack greater than 10 degrees, as indicated in Fig. 2-3(b).

Wind tunnel test results of a preliminary return booster configuration indicate subsonic  $(L/D)_{\max}$  equal to 7.0 at approximately 6-deg angle of attack. Hypersonic estimates show maximum  $L/D$  to be in excess of 2.0, with stable pitch characteristics beyond 60-deg angle of attack. The return booster wing is being configured to provide a stable transition between the hypersonic 60-deg entry and the subsonic cruise attitude at  $(L/D)_{\max}$ . This stable transition can thus provide supersonic gliding flight for additional range (thereby reducing subsonic requirements and vehicle weight).

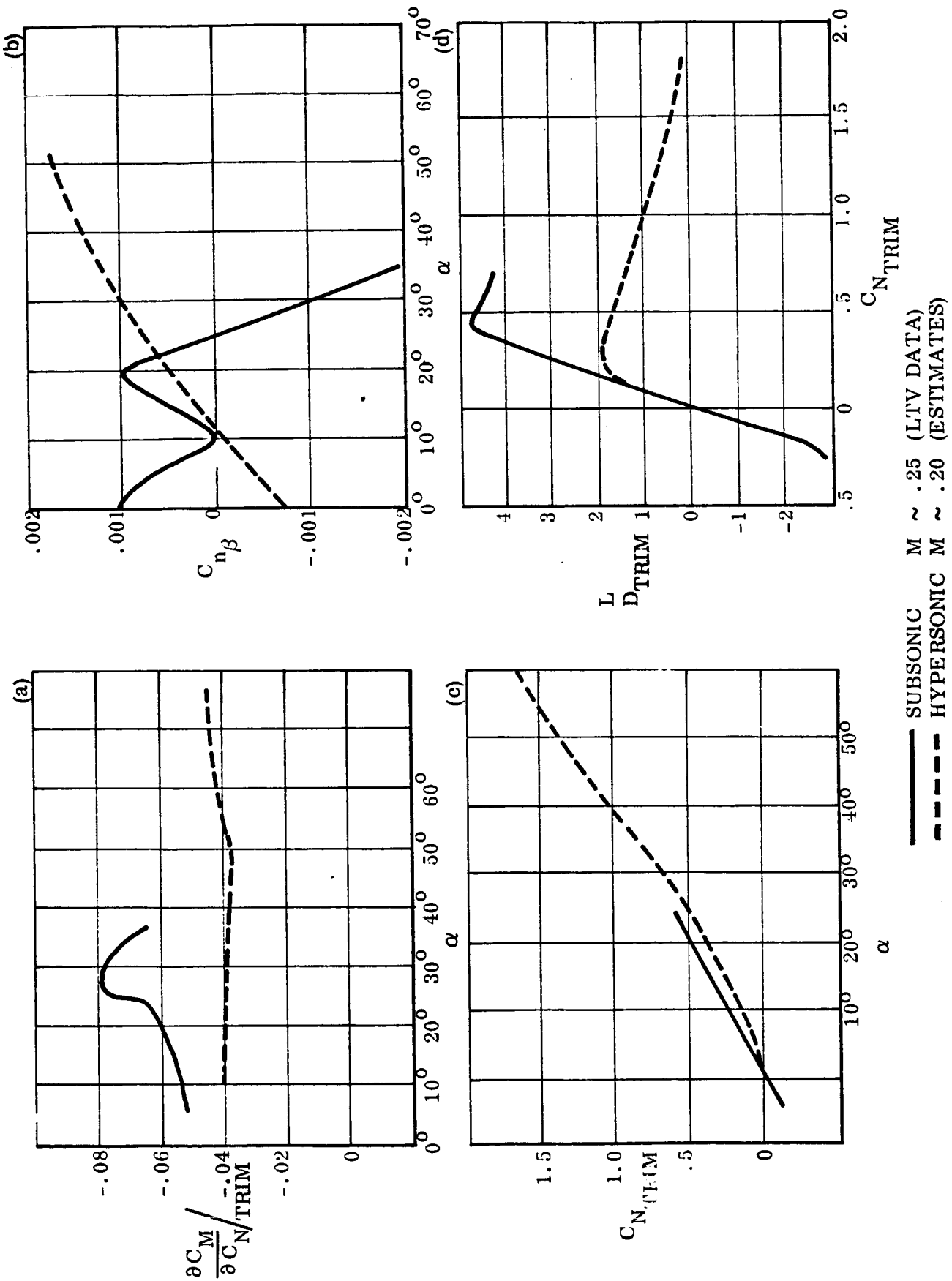


Fig. 2-3 Orbiter Stage Aerodynamics

### Aerothermodynamics

Ascent and entry thermal environments experienced by the orbiter and the booster have been computed. The orbiter entry trajectory provides a cross-range capability in excess of 1600 nm.

Figure 2-4 shows temperature histories for the booster wing tip leading edge stagnation line and for the wing upper and lower surfaces at a location 5 feet aft of the leading edge. The stagnation line peak temperature of  $1650^{\circ}\text{F}$  is the maximum experienced by the booster, with the exception of the base area. Peak temperatures on the upper and lower wing surface are 650 and  $1200^{\circ}\text{F}$ , respectively.

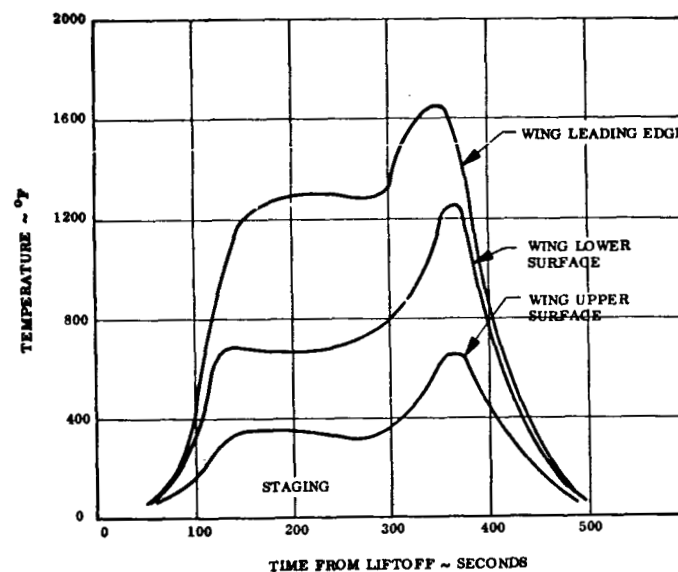


Fig. 2-4 Typical Ascent Temperatures

Shown in Fig. 2-5 are temperature histories at four lower surface locations for the orbiter, based on a 25-deg angle of attack entry trajectory, which generates 1600-nm cross range. Peak temperatures are  $2200^{\circ}\text{F}$  on the lower surface,  $2730^{\circ}\text{F}$  on the nose cap,  $2070^{\circ}\text{F}$  on the body leading edge,  $2200^{\circ}\text{F}$  on the fin leading edge, and  $1000^{\circ}\text{F}$  on the leeward surface.

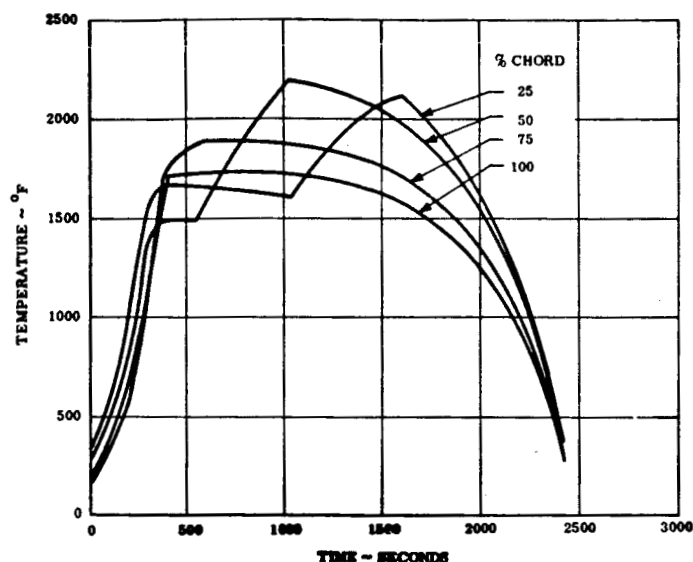


Fig. 2-5 Lower Centerline Temperature Histories for Reference Entry Trajectory

The percentage of surface area that experiences various peak temperature levels for the orbiter and the booster is given in Table 2-2. Less than 1 percent of the surface area experiences temperatures above 2500°F for the orbiter and 1500°F for the booster.

Table 2-2

PERCENTAGE OF SURFACE AREA FOR VARIOUS TEMPERATURE RANGES

Temperature Range	Booster	Orbiter (various cross ranges)			
		0 nm	500 nm	1000 nm	1500 nm
below 200°F	22	0	0	0	0
200 - 500	20	0	0	0	0
500 - 800	36	34	33	33	32
800 - 1500	22	11	12	13	13
1500 - 2000	0.3	25	27	28	30
2000 - 2200	0	30	28	26	25
2200 - 2500	0	0.3	0.4	0.4	0.4
2500 - 3000	0	0	0.1	0.1	0.1
over 3000	0	0	0	0	0

## 2.3 STRUCTURES, MATERIALS, AND THERMAL PROTECTION

Structural design criteria and the thermal load environment data were used to select candidate materials for the preliminary weight estimates. The results of an evaluation of candidate thermal-structural concepts and materials were used to define the structural arrangements.

### Structural Design Criteria

The basic structural design criteria used to govern the preliminary design of all the vehicle structural components provide a set of design conditions, requirements, and objectives to ensure that the structural components achieve acceptable compatible structural integrity. The structure is designed to survive the critical loading conditions and thermal environment on a specified number of missions with a minimum of structural refurbishment.

The following ultimate factors of safety were used:

- General structure 1.5
- Pressurized compartments 2.0

### Load Environment

Preliminary shear and moment loading distributions have been determined for use in establishing critical load paths and in sizing structural elements.

Maximum accelerations used for these load determinations are those established by the structural design criteria documents and by performance trajectory constraints. Conditions for which loading distributions have been determined are as follows:

- Maximum airload bending condition (maximum  $a_q$ ) occurring during ascent phase of flight
- Landing approach loading condition (subsonic maneuver) resulting from the 2.5-g low-altitude maneuver, based upon the requirements of MIL-A-8862
- A 2.0-g taxi condition with jet fuel tanks full

## Materials

The materials used for current weight estimates are summarized in Table 2-3. The choice of materials is based primarily on the expected thermal environment.

(See Table 2-2.)

The metallic heat shields and LI-1500 lightweight rigid insulation are leading candidates for thermal protection. A fiberglass-reinforced silicone elastomer ablator is considered as a backup system.

As indicated in Table 2-3, TD-NiCr is being considered for heat shield applications to 2200°F. While TD-NiCr has a short time capability to 2400°F, Cb-752 will be considered for ranges from 2200 to 2500°F for prolonged-temperature designs.

Table 2-3  
MATERIALS AND PREDICTED TEMPERATURES (°F)

Surface	Orbiter			Booster		
	Forward	Center	Aft	Forward	Center	Aft
Body Upper Heat Shield	700 to 1100 René 41	600 to 1000 René 41	500 to 1000 René 41	650 Ti	600 Ti	500 Ti
Body Lower Heat Shield	2000 to 2200 TD-NiCr	2000 to 2200 TD-NiCr	1800 to 2000 TD-NiCr	1200 René 41	1000 René 41	800 René 41
Nose	2750 Ta-10W	-	-	1450 René 41	-	-
Fin Leading Edge	-	-	2200 TD-NiCr	-	-	1650 René 41
Wing/Body Leading Edge	-	2080 TD-NiCr	2080 TD-NiCr	-	1650 René 41	1650 René 41
Wing Load-carrying structure	-	-	-	650 Ti	600 Ti	600 Ti
Wing Lower Heat Shield	-	-	-	1250 René 41	1200 René 41	1100 René 41
Primary Structure and Tanks, except for Wing of Booster	150, 2219-T87 Aluminum					

- Note: 1. LI-1500 also considered for heat shield  
 2. Cb-752, 2200°F to 2500°F (if required)  
 3. Ablator - backup heat shield

### Thermal Structural Concepts

Both the orbiter and the booster are of state-of-the-art aluminum primary structure, except for the wing of the booster, which is titanium. Any of three thermal protection systems listed below can be employed interchangeably:

- Metallic heat shields with internal insulation
- LI-1500 (an LMSC-developed lightweight rigid insulation on the external surface)
- Ablative heat shields (backup)

The use of interchangeable thermal protection systems presented above entail minimum development risks. During early vehicle flights, good temperature data are needed; this requires the use of a radiative (nonablative) heat shields. Existing metallic heat shield materials or LI-1500 can be used for early test flights. However, if temperatures are higher than anticipated for the fully operational flights, ablators are available for use on local areas of the vehicle. The various heat shields can be made interchangeable without greatly affecting the aluminum primary structure. The preferred metallic heat shield is a large corrugated panel with multiple clip supports.

Initially various thermal protection systems employing both passive and active systems were studied. Passive systems provide sufficient thermal insulation to limit the maximum structure temperature to an acceptable value. The following passive system concepts were evaluated:

- Felt-like high-temperature insulations, such as dyna-flex and microquartz in conjunction with metallic heat shields
- LI-1500
- A fiberglass-reinforced silicone elastomeric ablator ( $\rho = 20 \text{ lb/ft}^3$ )

The LI-1500 and metallic heat shield concept were also evaluated in conjunction with a closed-loop active cooling system. In all cases, orbiter internal structure was assumed to have a design maximum temperature of  $150^\circ\text{F}$ . Heating calculations were based on the  $L/D = 2$  spacecraft and maximum cross-range entry trajectory.

Because of the large potential saving in insulation weight, two approaches to alleviate the effects of post-touchdown heating were considered. One is to cool during low-speed flight with either ram air cooling or engine-bleed air; the other is to use a ground cooling system after landing. Analysis indicates that the use of ram air and ground cooling reduces the required LI-1500 thickness from greater than 5 inches to approximately 3 inches at the maximum heating lower surface location.

Shown in Fig. 2-6 are comparisons of the candidate thermal protection system weights (exclusive of the structure weight common to all systems). The ablator weight is based on an assumed 20 lb/ft<sup>3</sup> partial depth ablator with a bond-line temperature of 600°F. A 12-lb/ft<sup>3</sup> rigid insulator is used between the ablator and the structure.

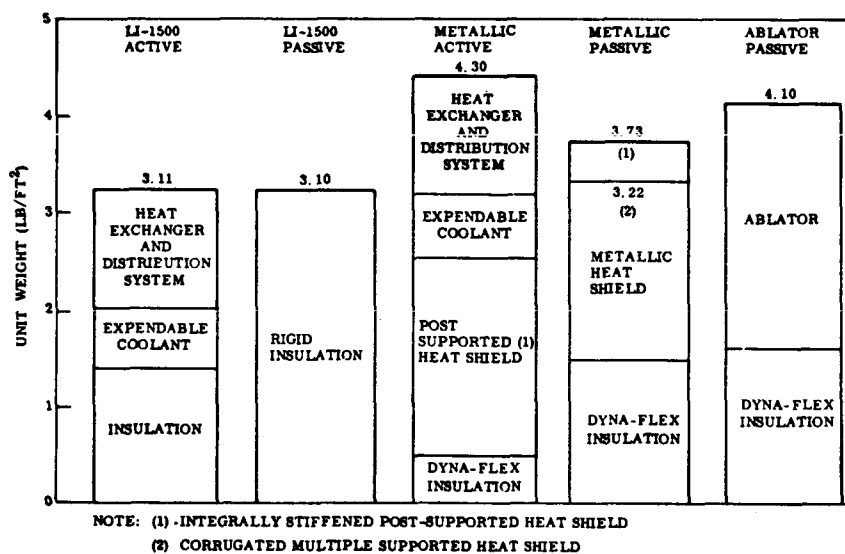


Fig. 2-6 Thermal Protection System Comparison (Orbiter)

Recent studies indicate that large corrugated heat shields with multiple clip supports are lighter in weight than post-supported integrally stiffened heat shields. The corrugated heat shield (Fig. 2-7) is mounted with a multiple clip arrangement through a glass rock insulator to the primary aluminum structure. Corrugation amplitude is one-tenth the corrugation pitch with a flat provided between corrugation arcs to enable attachment of the continuous support clip. Mechanical fasteners and resistance spot welding are used to attach the TD-NiCr and René 41 corrugated heat shields.

Blanket type insulation (dyna-flex and microquartz) is packaged between the corrugation shield and the structural panel.

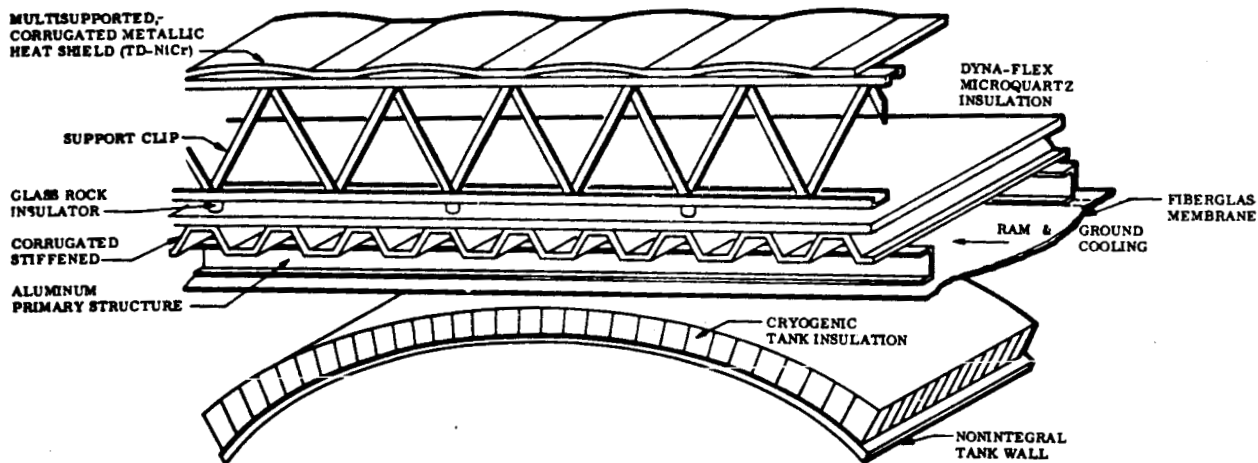


Fig. 2-7 Lower-Surface Metallic Heat Shield (Orbiter)

The LI-1500 material system shown in Fig. 2-8 is being considered as the outer surface thermal protection system for the vehicles in areas where the temperatures are 2500°F or less. The LI-1500 material protects the primary load-carrying structure and is subjected only to its own inertial loads and to air loads. The LI-1500 panels are bonded to the primary structural panels. Since the LI-1500 material has a very low thermal coefficient expansion, minimum external expansion joint are necessary.

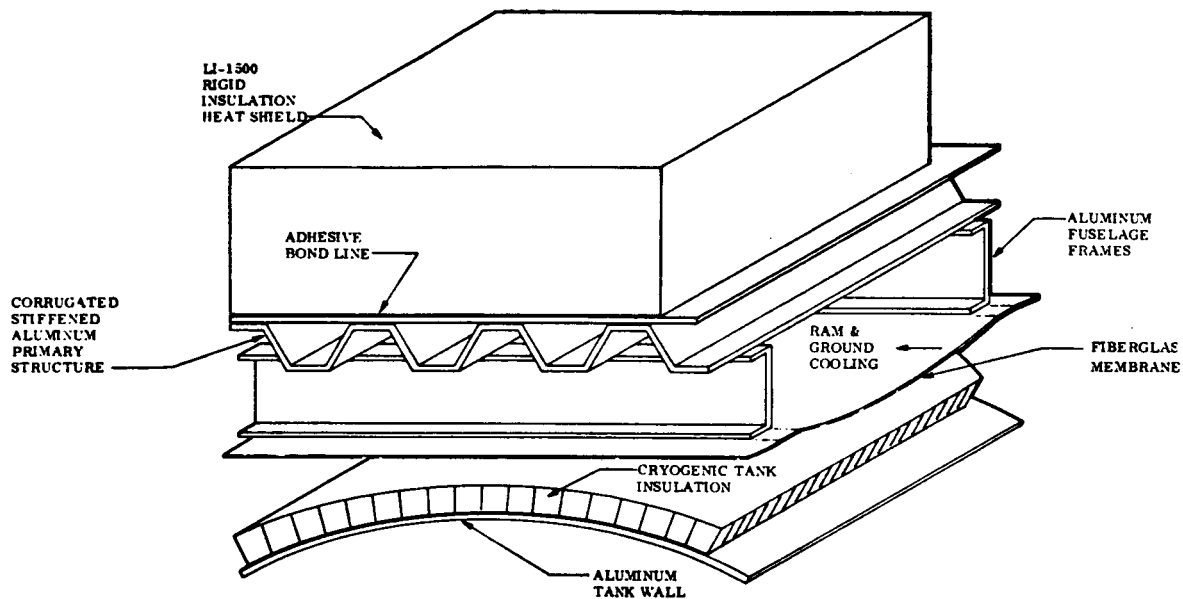


Fig. 2-8 LI-1500 Thermal Protection System (Orbiter)

Figure 2-9 shows the affect of heat shield weight on cross range for either metallic or LI-1500 heat shield concepts, which are competitive from the standpoint of weight. The heat shield arrangements shown in Figs. 2-7 and 2-8 are being considered for both the orbiter and the booster.

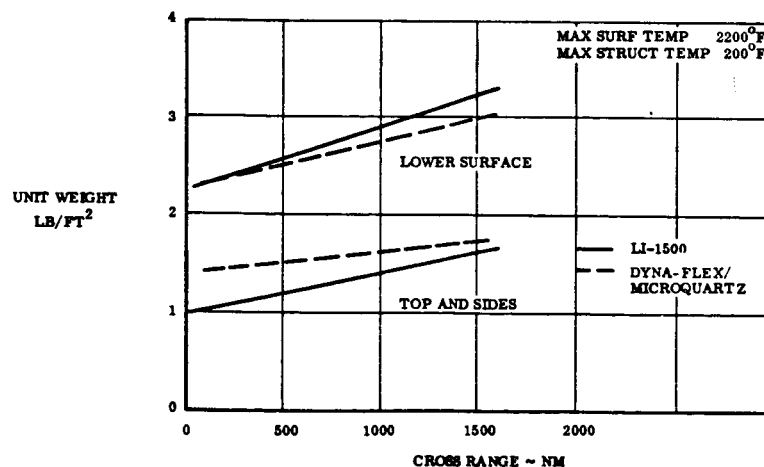


Fig. 2-9 Effect of Cross Range on Heat-Shield Weight

## Vehicle Structural Arrangement

The leading structural arrangements and materials presently being considered are as follows:

- **Booster**

Heat shielding: lower surface-René 41; upper surface-titanium

Load-carrying structure: fuselage-aluminum; wing-titanium

Load-carrying tanks (fuselage)

- **Orbiter**

Heat shielding: upper surface (René 41 or LI 1500, ablator, backup)  
lower surface (TD-NiCr or LI 1500, ablator, backup)

Load-carrying structure: aluminum

Nonload-carrying tanks

Orbiter. The orbiter structural configuration has a wedge planform shape with a triangular cross-section for the required lifting body characteristics. The body is basically of corrugated stiffened panel construction with intermediate supporting frames of aluminum construction to sustain the critical flight and landing loads. The propellant tanks are nonintegral to the fuselage structure and are designed by membrane stresses caused by the internal ullage and hydrostatic pressure.

The nose section of the fuselage is designed to withstand the collapse pressure and the body bending loads induced during the max  $\alpha q$  flight condition. Since the body bending loads are relatively small over this region, light-gage skins are used primarily to react the pressure loads to the closely spaced stringers and supporting frames.

Within the present planform design, based on the nonintegral tank concept, cylindrical and conical shaped  $LH_2$  and  $LO_2$  tanks are used to achieve the best use of the space available. Support of the propellant tanks to the primary fuselage structure is accommodated by circumferentially located tension link supports connected to the dome closure rings. These discrete support points reduce the heat flux to the primary shell and minimize the thermal stresses in the tank structure. The concentrated longitudinal forces induced in the fuselage structure are reacted out by longitudinal members and subsequent shear transfer in the skin.

At the aft fuselage tie-down point, the maximum booster acceleration loads at burnout are resisted by longitudinal members, which are tapered to minimize weight. The rocket engine thrust structure is integrated into this concept where possible to reduce the overall structural weight due to thrust loads.

Booster. The first stage comprises two basic structures, similar to conventional aircraft; a cylindrical fuselage, housing the  $\text{LO}_2$  and  $\text{LH}_2$  propellant; and a wing box. The propellant tankage (in the fuselage) is designed as primary load-carrying structure to minimize vehicle weight. The  $\text{LO}_2$  tank (2219-T87), being located in the forward section, is tied structurally to the  $\text{LH}_2$  tank (2219-T87) with a conventional missile interstage ring-stringer stiffened shell.

The  $\text{LO}_2$  tank is ring stiffened only and is designed principally for a combination of internal ullage and hydrostatic pressure. The  $\text{LH}_2$  tank is designed by a combination of internal pressure, body bending, and axial boost loads. To provide for axial stiffness for the various flight and ground conditions, integrally machined T-shaped longitudinal stiffeners are provided (internally). Channel-shaped rings are spaced about 50 inches apart.

At the aft skirt section, the engine thrust structure is built up from a space frame concept, with the second stage shear tie structure integrated into the space frame. Thrust load concentrations at the aft end of the  $\text{LH}_2$  shell are sheared out by providing tapered longitudinal internal straps attached to the T-stiffeners.

The wing load-carrying structure, of titanium, consists of beaded cover panels, stiffened in the spanwise direction. Channel rib caps carry the chordwise loads. This panel concept was developed by LMSC under NASA Langley Research Center Contract NAS1-7573 ("Hypersonic Cruise Vehicle Wing Structure").

A René 41 heat shield (of a design similar to that of the fuselage heat shield) and insulation are used on the lower surface to restrict the internal titanium structure to  $600^\circ\text{F}$ . (LI-1500 is also considered for the lower surface heat shield.) While the upper surface primary structure of titanium does not exceed  $600^\circ\text{F}$  during flight, a lightweight titanium heat shield is used as an aerodynamic fairing (since the beaded primary panel runs in a spanwise direction).

## 2.4 PROPULSION SYSTEMS AND PROPULSION INTEGRATION

Propulsion requirements are met by three systems — primary, reaction control, and subsonic cruise. In addition, an auxiliary power system provides hydraulic power for reentry and landing.

### Primary Propulsion Systems

This primary propulsion system for each stage consists of oxygen-hydrogen, continuously throttlable rocket engines (supported by a propellant storage system) and thermal protection, plumbing, pressurization, and propellant management subsystems. System characteristics are given in the following table.

Table 2-4  
SYSTEM CHARACTERISTICS

	Number of Engines	Rated Thrust (SL)	Propellant Weight (lb)
Orbiter	2	560,000	743,350
Booster	11	560,000	1,515,700

The engine selected is the Pratt & Whitney XLR-129, a reusable high-pressure, staged-combustion engine with either a fixed or a two-position bell nozzle, which permits operation of the engine for boost from sea level and for orbital operations with optimum nozzle area ratios for each phase of the launch trajectory.

The orbiter is equipped with a number of cylindrical or conical tanks, thermally decoupled by low-conductivity standoff structure and insulation. One  $\text{LH}_2$  storage tank is located forward and several in the aft area of the orbiter, depending on the payload configuration. All LOX storage tanks are mounted in the forward area of the orbiter. The LOX and  $\text{LH}_2$  storage tanks drain into LOX and  $\text{LH}_2$  sump tanks, from which the propellant is fed to the engines through manifolds. The sump tanks are insulated with multilayer superinsulation for long-term storage of propellant in orbit. During ground hold, the insulation is purged with helium.

The booster is designed to have insulated cylindrical load-bearing propellant tanks, integrated into the fuselage structure. The LOX tank (located forward) and the fuel tank (aft) together occupy most of the fuselage volume. LOX flows from the tank via a single line to a manifold located just forward of the engines. The manifold distributes the flow to each of the 11 engines. The fuel is distributed directly from the tank sump to each engine via individual lines.

For automatic control over propulsion system checkout and inflight system surveillance, propulsion subsystem instrumentation will be incorporated in such a manner that information on propellant loading, propellant gaging, and engine operation will be continuously monitored and controlled by a computer. As part of the engine design, a malfunction model will be established and computer routines developed for detecting engine malfunctions and for initiation of corrective action.

#### Reaction Control Propulsion System

The cryogenic propellant combination of oxygen and hydrogen is employed in the baseline reaction control system (RCS). Propellants are stored as liquids in the spacecraft main propulsion system tanks, but they are supplied to the RCS as gases.

To perform its multiple function, the RCS must meet the following requirements:

- Three-axis slewing about any axis
- Three-axis attitude control
- Three-axis translation along any axis

The minimum impulse bit and the thrust levels for attitude hold, maneuvering, and translation required for the orbiter are listed in the following table.

Table 2-5

## IMPULSE BIT AND THRUST LEVEL REQUIREMENTS

Limit Cycle		Function	
Impulse Bit (lb-sec)	Thrust (lb)	Attitude Maneuvers Thrust (lb)	Translation Thrust (lb)
10 - 12	100	2160 to 895	3480 to 2760

The RCS consists of 30 thrusters. For limiting cycling operations, small thrusters (100 pounds thrust) are pulsed individually to minimize pulse impulse and propellant consumption. For attitude maneuvers and translation, the large throttlable thrusters (3500 pounds maximum thrust) are fired as required at the selected thrust level.

Preliminary performance characteristics, shown in the following table, are based on information supplied by engine manufacturers.

Table 2-6

## RCS PERFORMANCE CHARACTERISTICS

	Limit Cycle	Maneuvering/Translation
Thrust (lb)	100	3500/875
Specific Impulse (sec)		
Steady State	410	420
Pulsing	336	-
Minimum Impulse Bit (lb-sec)	10 - 12	-
Mixture Ratio	5	5
Chamber Pressure (psia)	300	300
Thruster Area Ratio	40:1	40:1
Thruster Weight (lb)	8	50
Thruster Length (in.)	10	35
Thruster Diameter (in.)	3	18
Propellant Feed Pressure (psia)	500	500
Propellants	GO <sub>2</sub> /GH <sub>2</sub>	GO <sub>2</sub> /GH <sub>2</sub>

### Subsonic Cruise Propulsion System Summary

The subsonic cruise propulsion system (SCPS) provides the thrust to meet the powered approach, ferry, and go-around capability for both the orbiter and the booster. The SCPS requirements are shown in the following table.

Table 2-7

#### SCPS REQUIREMENTS

Parameter	Orbiter	Booster
Number of Engines	4	4
Engine Thrust Level (sea-level static) (lb)	46,000	63,500*
Total Thrust (sea-level static) (lb)	164,000	254,000
Intended Use (standing, ferry)	Both	Both
Subsonic Cruise for Cross-Range Capability	15 min plus go-around	—

\*Includes a 10 percent factor for climb to 10,000-ft cruise altitude

Two fuel tanks are incorporated in the basic airframe for powered approach and go-around for both the orbiter and the booster. For ferrying purposes, an auxiliary tank is installed in the cargo compartment of the orbiter.

The SCPS engines for the orbiter will be retracted throughout the on-orbit and launch mission phases. After subsonic entry flight conditions have been reached, the engines will be deployed and air started by windmilling. The SCPS engines for the booster are mounted so that no deployment or retraction is required.

As a result of tradeoff analyses, a turbofan with a bypass ratio of 5 has been defined as the baseline engine for the subsonic cruise propulsion system.

Auxiliary Power System

Auxiliary power systems (APS) are required for both the orbiter and the booster. Preliminary power requirements of the orbiter are presented below.

Table 2-8

## ORBITER APS POWER REQUIREMENTS

Mission Phase	Duration (min)	Power (hp)	Energy (hp-hr)
Entry	40	15	10
Landing	12	50	10
Total	52	—	20

The APS will consist of four turbine-driven power units, employing hydrogen and oxygen as reactants at a one-to-one mixture ratio. Normal power for peak demand is supplied by three of the units, with the fourth on idle standby. For normal power demand, one unit will be operated with one on idle standby. The power unit on the orbiter is the power source for the hydraulic system. On the booster, the power unit may supply electrical power as well as hydraulic power.

Preliminary APS characteristics are presented in the following table.

Table 2-9

## AUXILIARY POWER SYSTEM CHARACTERISTICS

APU Type	Chemically fueled, turbine driven
Number of Units	3 on-line, 1 standby
Status	To be developed
Energy Requirement	20 hp-hr
Propellant Weight	60 lb
Hydrogen Weight	30 lb
Oxygen Weight	30 lb
Auxiliary Power Unit Requirements	
Power	17 hp
Reactants	Hydrogen and oxygen (1:1), stored cryogenically
Specific Propellant Consumption	3 lb/hp-hr
Life	200 hr

## 2.5 AVIONICS

Avionics includes all electronic systems used in operating the vehicle and the data management equipment. Data management may be considered as a base from which an integrated system recommendation will evolve in that it includes onboard checkout and fault-isolation capability.

### Electrical Power

The orbiter contains fuel cells for electrical power and a standby battery system for deorbit, entry, and landing power. The booster is powered by primary batteries. Both stages have provision for power to be supplied by alternators, driven by the main turbojet thrusters during operations in the atmosphere. The high mechanical power requirements associated with aerodynamic control surfaces are supplied by a chemical, turbine-driven, hydraulic system during entry and landing phases. Average electrical power requirements range from 3 kw to 5 kw.

The baseline distribution system is composed of two 28-v unregulated dc buses and two 115-v ac buses. Solid-state inverters supply 3-phase, 400-cps power to the ac buses. Lightweight, high-efficiency, power control and distribution systems will be required because of the wide dispersment of electrical user equipment throughout the vehicle. It is estimated that power control and distribution will make up at least 75 percent of the power system weight. Flat conductor cable and connectors are recommended for power and signal distribution, and high voltage (50 to 100 v) inverter and converter hardware is being considered for power distribution.

### Guidance, Navigation, and Control

This system must provide position determination, trajectory control, attitude control, and stability augmentation during the ascent, booster return, orbital operations, re-entry, and approach and landing. It must be capable of completely autonomous operation after prelaunch data readin except for a landing guidance data link. Backup provisions should include state vector update by Navsat data link and pilot control during all mission phases.

The system consists of six subsystems.

Sensor/Processor (Vehicle State) Subsystem. Composed of all state-sensing instruments and required data processing equations, this subsystem provides the complete description of the state of the vehicle with respect to the outside world during all mission phases. Typical state parameters are vehicle position, velocity, attitude, Mach number, angle of attack, and skin temperature.

The baseline subsystem consists of a strap-down inertial measurement unit assembly, star sensor, horizon sensor, radar altimeter, skin temperature thermocouples, rate gyro package, and air data sensor. The number of individual equipment packages required will be established by an analysis of system reliability and safety requirements.

The booster system will require only the inertial measurement unit, computer assembly, rate gyro package, air data sensor, and landing aids.

Guidance Subsystem. There is no guidance subsystem in the conventional sense. Hardware requirements are met by the state sensing instruments and the computer subsystems. Software requirements are met by computer equation routines for processing vehicle state data to produce trajectory constants before launch and in-flight steering orders as well as ignition, cutoff, and throttling orders to the flight control subsystem and pilot display.

Stabilization and Control Subsystem. This subsystem is physically composed of flight control drive electronics and electrohydraulic servo actuators. Equations are used to process guidance orders and vehicle state data to produce actuation signals and pilot display data.

Rendezvous and Docking Subsystem. Composed of sensors (e.g., laser transmitter/receiver) and associated data processing equations, this subsystem provides relative state data and steering and RCS thrusting orders to the flight control subsystem and pilot display data.

Approach and Landing Subsystem. Composed of sensors (e.g., radar transponders) and associated data processing equations, this subsystem provides relative state data and steering and thrusting orders to flight control subsystem and pilot display data.

Computer Subsystem. Composed of computer, computer controls, computer software, and pilot display data, this performs all data processing. It may be part of a federated or integrated avionics computer system.

### Communications

Communications requirements have been investigated for the orbiter but not specifically for the booster, since it does not impose additional requirements except for increased data rates and the need for communication between stages.

Performing as a data link, communications subsystems provide critical data through the data management subsystem to the ground and to the space station as a backup to information supplied to the crew. As a command link, it automatically transfers data from the ground or from the space station to the orbiter as a backup to the crew. Inter-communication on board the orbiter is also provided by the communication subsystem.

Orbiter-to-ground communication is a requirement during all mission phases except reentry, where uninterrupted communication through the plasma sheath is desirable but not a firm requirement. Since the orbiter is to be autonomous, transmission of data and control will be minimal and for backup purposes only. Down-link data rates during ascent, orbit, and reentry phases are estimated to fall below 16,000 bits/sec, 17,000 bits/sec, and 37,000 bits/sec, respectively. The up-link command data rate is 1,000 bits/sec, and 3 kHz voice communication is provided throughout. Orbiter-to-ground communication via satellite is also a requirement.

The communication link for long range will be by a low-power transmitter directly to the ground when feasible, with additional power amplifier stages to go to the communication satellite when necessary. The receiving function will have regular front ends for the ground communication, with the addition of cryogenic front ends for satellite links. The antennas will be flush mounted on both top and bottom of the vehicle, with omnidirectional capability to the ground and electronically controlled directivity for communication

with the satellite. This will allow the use of minimum power when possible. The frequencies to be used on the long-range link will be compatible with the ground and satellite frequencies that are to be used in this time period.

For short-range communication during ascent, rendezvous and docking, and landing, a UHF frequency link will be used. This will minimize the number of configurations and still meet the FAA control requirements during the landing phase.

#### Control/Display

The control/display subsystem will be designed to simplify the presentation of information to the crew and to reduce the crew's workload for status monitoring and for control of the vehicle and its subsystem. The extensive use of grouped, individual, dedicated displays is not considered applicable for the Space Shuttle. Instead, a very flexible, programmable display system, incorporating wide-angle cathode ray tubes for simultaneous display of many parameters by multifunction techniques, is considered to be more suitable.

A preliminary list of test points, parameters, ranges, and resolution requirements for orbiter functions indicates approximately 500 display modes and 300 control modes.

The use of multiplexers to accept analog inputs from a variety of slow-moving (low-frequency) transducers vastly reduces the amount of wire and cable required to run signals throughout the vehicles.

All manipulation of values will be in terms of digital values, an approach that lends itself to modern digital processing techniques.

Keyboards or similar crew-manipulated input devices will be used to select particular display patterns on programmable display devices. State-of-the-art devices will be employed to allow use of a very simple keyboard for literally hundreds of configurations or modes.

A keyboard control device used to evaluate the mode selected enables the single, simple keyboard to provide for great numbers of selections. A further function of keyboard control is to provide information back to the keyboard operator, describing the selections made.

A displayed parameter selector provides the function of manipulating the remote (and nonremote) multiplexers and gates, signal conditioning, and digitizing equipment so that the values of the parameters to be displayed will be available to the display control processor.

A display control processor must be provided to accept the selected display modes from the crew keyboards, the input values of the parameters to be displayed, and characters and symbols from the generation equipment. It must contain or be intimately associated with a display refresh memory (if the selected programmable display devices require refreshing) and must also contain a character, symbol, and vector generator.

#### Data Management

Two sets of requirements must be considered in arriving at the data management subsystem design:

- Requirements for acquiring vehicle subsystem data and for applying pre-determined rationale to request an automatic response or, alternatively, to present information to an operator for evaluation and for initiation of a desired action
- Earth-based requirements attendant to an operational vehicle with a highly mechanized data management capability

The requirements for safety in manned flight operations dictates the need for redundancy in vehicle subsystem designs and the need for intact abort, much as is the case of commercial and military aircraft. Onboard checkout, fault isolation, and warning of an abort situation must therefore be provided within the vehicle's data management subsystem. In addition, the need for decreased maintenance time on the ground, i. e., rapid turnaround, also establishes the need for onboard checkout and fault-isolation capability for maintenance purposes. With such capability on board the vehicle, near-autonomous operation becomes an attractive possibility.

The successful application of an integrated data management subsystem to the Space Shuttle system design will accomplish several results:

- Simplify the crew tasks by assimilating large quantities of data through performing rapid calculations and logic decisions, monitoring and interpreting the operating control system signals simultaneously, and providing only pertinent data to the operator as a function of mission phase.
- Support attainment of the maximum flight utilization of the shuttle for minimum support costs over the projected system life by performing the function of on-board checkout and fault isolation. A secondary benefit is automation of a significant level of historically important failure record data and flight logs and collection of trend prediction data in machine compatible format.
- Provide the hardware for satisfying the demands of autonomous operation represented by self-contained countdown (the function of range safety monitoring excepted); internal control of external fueling facilities; maintaining realtime surveillance of vehicle integrity, including abort warning; providing mission support in the form of time-line instructions, special semiautomated check lists for tasks such as rendezvous/docking, rescue missions, and simplified technical order material for on-station troubleshooting.

Attainment of a truly integrated avionic subsystem may well be an evolutionary process during the operational life of the Space Shuttle, provided that a viable growth concept is devised on the basis of initial definition of the final integrated system to be achieved. The attractiveness of this solution lies in the ability to minimize the technical risk of the initial acquisition, yet retain the ability to achieve the benefits of integrated avionics by evolutionary steps in subsystem areas that offer the greatest benefits. Such an evolutionary approach must be compared with the approach of maximum integration initially, contained only by available technology. The latter represents an initially greater technical risk and cost but, if properly designed, requires little or no modification, providing immediate benefits (in terms of increased reliability, decreased weight, and decreased power) and potential long-term cost savings.

A functional description of the subsystem is represented in the following block diagram.

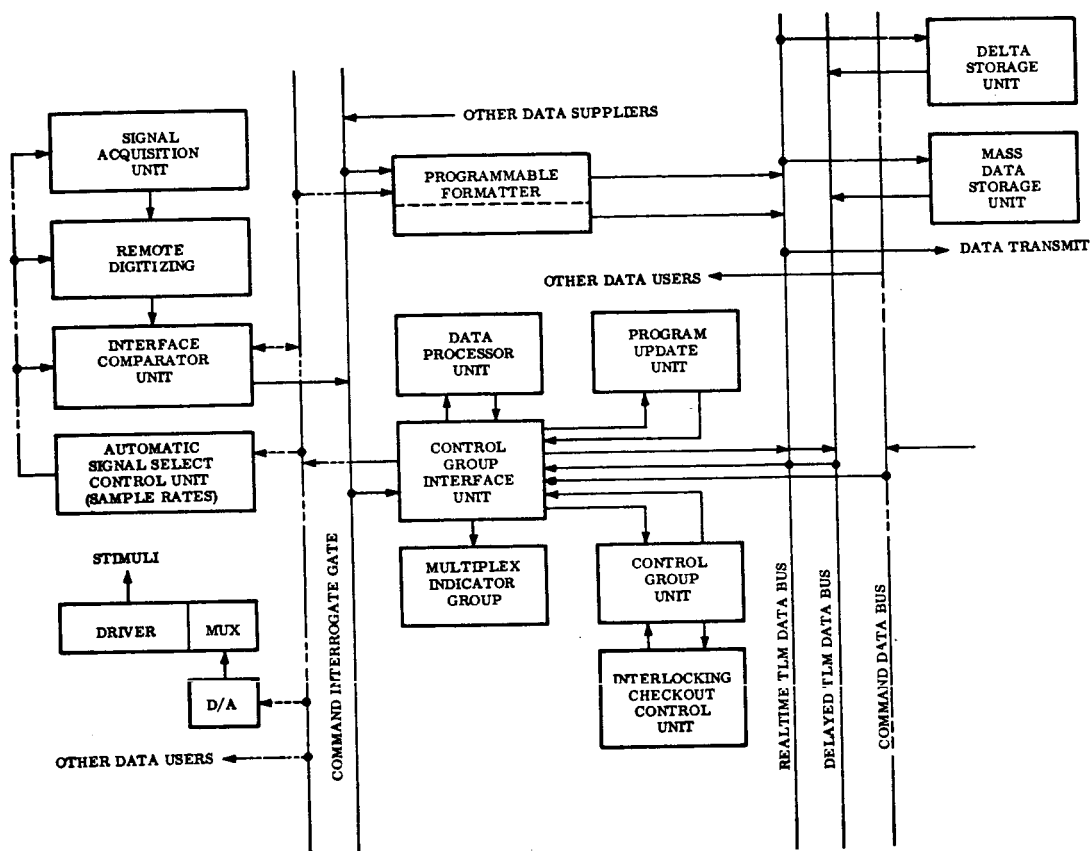


Fig. 2-10 Data Management

## 2.6 PERFORMANCE AND FLIGHT MECHANICS

The combined ascent/reentry trajectory profile of the Two-Stage reusable Space Shuttle is shown in Fig. 2-11.

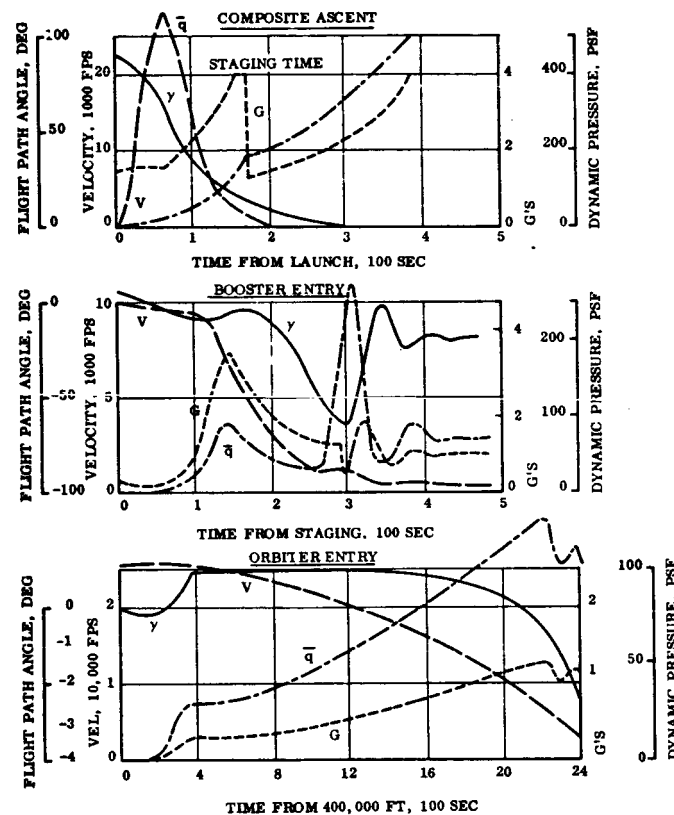


Fig. 2-11 Ascent/Reentry Trajectory Characteristics

### Ascent

Passenger levels should not exceed three. However, structural requirements are derived from the 4-g reference trajectory shown, which is used for cargo flights. The mission velocity capability incorporated is set to cover both the 4-g and 3-g ascent. After reaching the acceleration limit, the g level is maintained by continuous throttling. Stage separation occurs after about 3 minutes at an altitude of 240,000 ft and 160 nm downrange at a dynamic pressure of less than 40 psf. The orbiter burns for an additional 3.5 minutes until injection into the 45 by 100 nm transfer orbit.

### Booster Reentry and Flyback

After separation, the booster coasts briefly and reenters the atmosphere with 3.6-g maximum deceleration. By the time the flyback cruise altitude of 10,000 ft is reached, the booster will have completed a descending turn with 210-deg azimuth change. Entry angle of attack is held at 60 deg until a 60,000-ft altitude is reached and then lowered to 15 deg. The booster entry time is 6 minutes, and the flyback range is 370 nm. Flyback cruise time is on the order of 2 hours.

### Orbiter Reentry

About 40 minutes after retrofire, the orbiter enters the sensible atmosphere (4000,000 ft) at about -1-deg flight path angle.

For the reference trajectory, the total entry time to 100,000-ft altitude is about 40 minutes and accelerations are limited to 1.25 g's.

At 45,000-ft altitude (Mach 0.7), the airbreathing engines are deployed for a power-assisted landing. The duration of the approach and landing phase is about 16 minutes, accounting for the possibility of a go-around.

### Reentry Heating

In order to maximize the portion of the entry trajectory above the heating boundary (defined by the flight profile that results in a specified constant lower surface temperature) at a given cross range, angle of attack, and roll angle, modulation is employed. Therefore the trajectory parallels the heating boundary for a longer period of time than a unmodulated equilibrium path could. A relatively deep first plunge with immediate maneuvering is made feasible by increasing the leading edge radius to approximately 3 ft so that the lower surface heating becomes the determining factor. Maneuvering is temporarily suspended during the closest approach to the heating boundary and is later resumed within the heating and acceleration constraints.

The entry corridor height is defined as the minimum vertical displacement between the wing's level equilibrium glide and the heating boundary. Its relationships to temperature and wing loading within the context of the aforementioned maneuvering philosophy

is given in Fig. 2-12. The reference entry trajectory presented earlier is based on a lower surface temperature limit of  $2200^{\circ}\text{F}$ . This trajectory is shown in Fig. 2-13 in the form of an altitude-velocity plot.

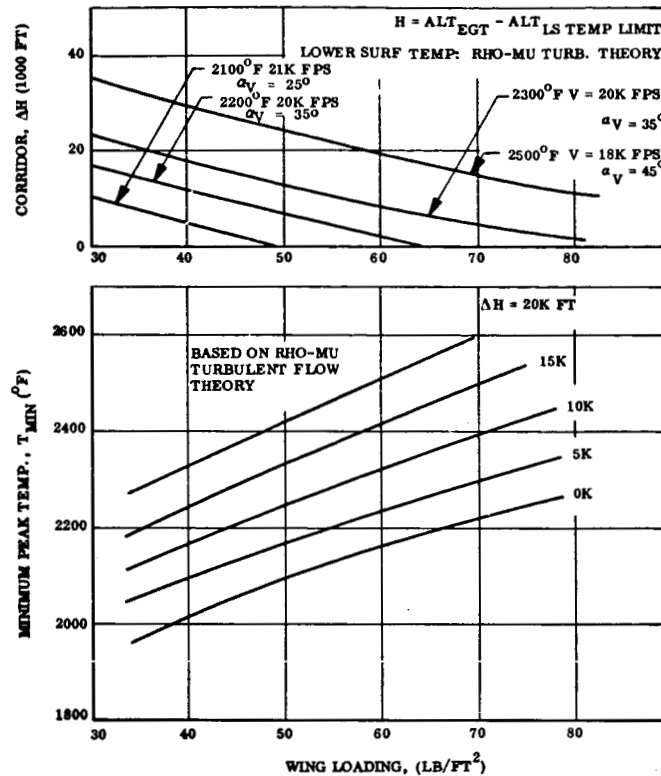


Fig. 2-12 Minimum Peak Temperature as a Function of Wing Loading

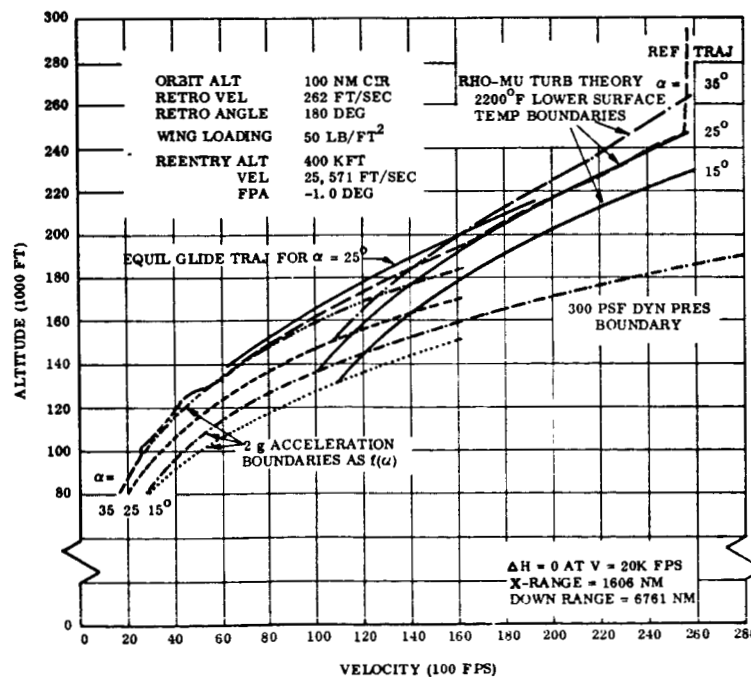


Fig. 2-13 Reference Trajectory Altitude-Velocity Profile  
2-32

### Approach and Landing

A landing profile, established as shown in Fig. 2-14, permits both an airliner-type powered landing and an unpowered emergency landing if the jet engines fail to deploy or start. During descent from reentry, the orbiter is established at a maximum L/D glide and is controlled by energy management techniques to arrive at the landing approach window (decision key). In a normal powered landing, a 360-deg descending turn is made to a 3-deg final approach. For an unpowered landing, a straight glide is made to the runway. Analysis shows that vehicle performance capabilities are suitable for either landing. Touchdown speed is 180 knots, with rollout distances on the order of 6000 to 7000 ft.

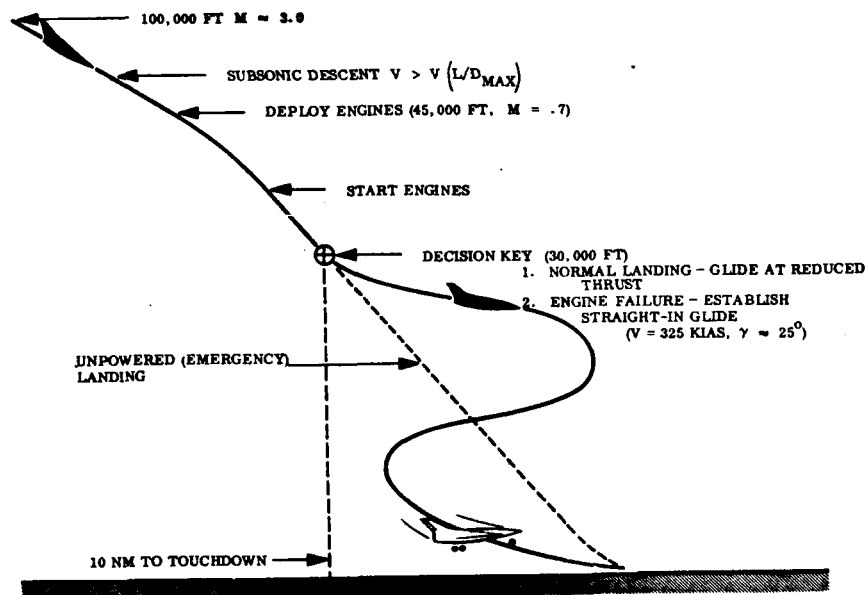


Fig. 2-14 Approach and Landing Profile

A go-around profile has been established, and analysis indicates engine thrust requirements of 87,000 lb for the spacecraft and 113,000 lb for the booster. An estimate of the orbiter self-ferry capabilities with these engines shows a limited ferry range of approximately 300 nm.



### Refurbishment

The onboard checkout system greatly simplifies refurbishment operations in that it facilitates fault detection, isolation, and status monitoring. Key refurbishment considerations are as follows:

- Design for automatic onboard checkout and status monitoring. Include appropriate transducers in mechanical, pneumatic, electrical, structural, and hydraulic systems for performance verification or malfunction detection.
- Provide hard points for handling large components, the orbiter, and the complete launch vehicle. Allow for standard ground shock and vibration environment (in most cases not a limiting item).
- Provide field joints to enable breaking down large items to transportable size.
- Design for maintainability. Locate limited life components so that they are accessible for ready replacement. Use a modular approach wherever possible.
- Incorporate startup provisions for the fuel cells in the electrical power system. Dummy loads may be required.
- Provide optical targets for rapid alignment checks during assembly and repair and at the launch pad.

Average normal refurbishment times are estimated to be ten shifts for the orbiter and seven shifts for the booster.

### Launch

Pad operations are significantly expedited by use of the onboard checkout system and minimum support crews for basic jobs such as vehicle installation, propellant loading, and countdown. The shuttle stages are mated horizontally in the Maintenance and Assembly Building, towed out to the launch pad, installed, and erected.

### Flight

Flight operations concern the ground support, crew functions, and vehicle operations for ascent, on-orbit, reentry, landing, and ferry modes. Consideration of mission spectra, rendezvous techniques, docking operations, cargo handling systems, and orbital tug operation are important in design of the operational vehicles and support systems.

### Landing Support Operations

Landing operations are based on automatic, hands-off, all-weather landings. Ground support operations include crew removal, spacecraft cooling, cargo unloading, propellant purging, protection of sensitive systems, and protection against fire and environmental factors.

### Ferry Support Operations

Ferry mode capability for all shuttle vehicle elements is provided by adding auxiliary JP-4 tanks to the basic vehicles and using the installed turbofan engines for horizontal takeoff and cruise.

### Operations Management

The Operations Management Center performs the administrative, scheduling, planning, logistics, record keeping, and control functions for all Space Shuttle operations.

### Facilities

Some of the facility considerations under study are as follows:

- Launch facility types, sizes, and locations
- Launcher types
- Erection concepts
- Maintenance and Assembly Building layouts
- Landing fields

### Safety

The safety approach is based on proven commercial and military aircraft safety methods and procedures. Key safety factors include the following:

- Operate a high-performance rocket vehicle with techniques used in military and commercial aircraft test and operation.
- Exploit potential afforded by advanced staging concepts, engines and sub-systems, and autonomous checkout and landing systems.
- Provide primary intact abort mode throughout development test, IOC, and operational phases. (Design to cope with emergencies.)
- Provide complete crew capsule escape backup during the development test phase.

- Adopt best practices of Apollo-11 for critical component factory acceptance.
- Adopt military/commercial fail operational-fail safe backups in vehicle subsystems, procedures, and operations.

The phases of flight safety development associated with increasing confidence gained in the system reliability and performance are as follows:

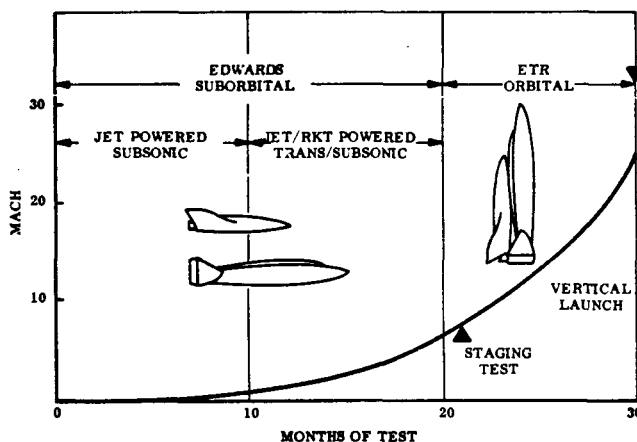
- Phase I - Development - provisions for major and catastrophic failures
- Phase II - Initial operations - provisions for major failures, not catastrophic
- Phase III - Fully operational - provisions for subsystem failures, not catastrophic

Throughout all three phases, spacecraft abort-to-orbit and return-to-earth intact are the primary abort modes; however, during the development phase, crew encapsulated escape will be provided as a backup to the primary abort modes.

## 2.8 TEST AND PRODUCTION

### Development Testing

Analyses have been performed to determine the gross effect of introducing the reusable vehicle concept on development testing. Results indicate that application of aircraft buildup flight test concepts would permit schedule compression and reduced emphasis on ground testing while maintaining acceptable test risk. Such a flight test concept is illustrated in Fig. 2-17. However, the problem of smooth progression through the various flight regimes must be fully investigated. Proposed testing includes a 30-month flight test program, including performance demonstration, which will be performed in parallel (schedule) with ground qualification of noncritical items. Ground qualification of critical items will be conducted prior to flight on an expedited basis.



**Fig. 2-17 Flight Test Concept**

Critical issues to be evaluated in the course of the combined ground/flight test program include:

- Determination of experimental flight stress and temperature buildup profiles for hypersonic flight to evaluate heat shield performance and structural dynamic loads
- Validation of RCS/aero control blending and basic control stability throughout subsonic, transonic, and supersonic regimes
- Normal vehicle staging and abort staging

The current accuracies of turbulent heating theory and other aspects of thermal prediction are such that peak reentry temperatures could be significantly higher than predicted; the uncertainty is approximately  $200^{\circ}$ . Considerable uncertainty will remain until the actual flight test of the vehicle. Extensive ground tests and subscale flight tests can reduce these uncertainties but not eliminate them. The buildup flight testing entails such a progressive approach.

The flexible heat shield design, which allows for the use of alternate materials (metallic, LI-1500, or ablators), makes possible a progressive approach in materials as well as in the flight test trajectories employed. In the initial flight testing, it will be desirable to use radiative (nonablative) systems so that reliability thermal measurements can be made. Initial trajectories will have to be carefully planned to assure avoidance of excessive temperatures. As much refinement of thermal prediction as possible will be done in suborbital flights that penetrate the lower regions of hypersonic flight. Uncertainties will, however, still dictate a conservative approach to the first orbital reentries, involving minimum banking (limited cross range) and angle-of-attack modulation to minimize temperatures.

### Reliability

The fail-operational concept of the reusable booster and orbiter dictates that the reliability of the systems approach unity in order to meet mission objectives. Furthermore, it is necessary that the design of the systems be such that serviceability and maintainability are high to meet the total operational availability.

To ensure the safety of crew and passengers, the following essential points must be recognized in all aspects of design and operation:

- Identification of all critical functions, events, equipment
- Assessment of failure modes and frequency
- Achievement of reliability through fail-safe methods of redundancy and failure effect containment
- Attainment of optimum, safe man-system interfaces
- Implementation of fault detection, identification, and correction

Crew survival probability analyses have been conducted for all flight phases with the aid of hazard logic block diagrams for each system under study. Semi-empirical probabilities of reliability have been assigned to critical events. Thus, for a given total mission reliability, it is possible to use the model to apportion reliability objectives for each mission phase. This, in turn, leads to the determination of reliability goals for individual subsystems needed to function during the relevant phase to ensure safe abort.

#### Manufacturing and Procurement

Particular attention in this area has been given to fabricability, assembly sequence, tooling, packaging, handling, and transportation. Because of the difficulty in handling and transporting large vehicles, a modular approach appears to be attractive. The modules would have hard interfaces to permit final assembly at or near the launch site. Typical manufacturing assembly spans for the orbiter are shown in the Fig. 2-18.

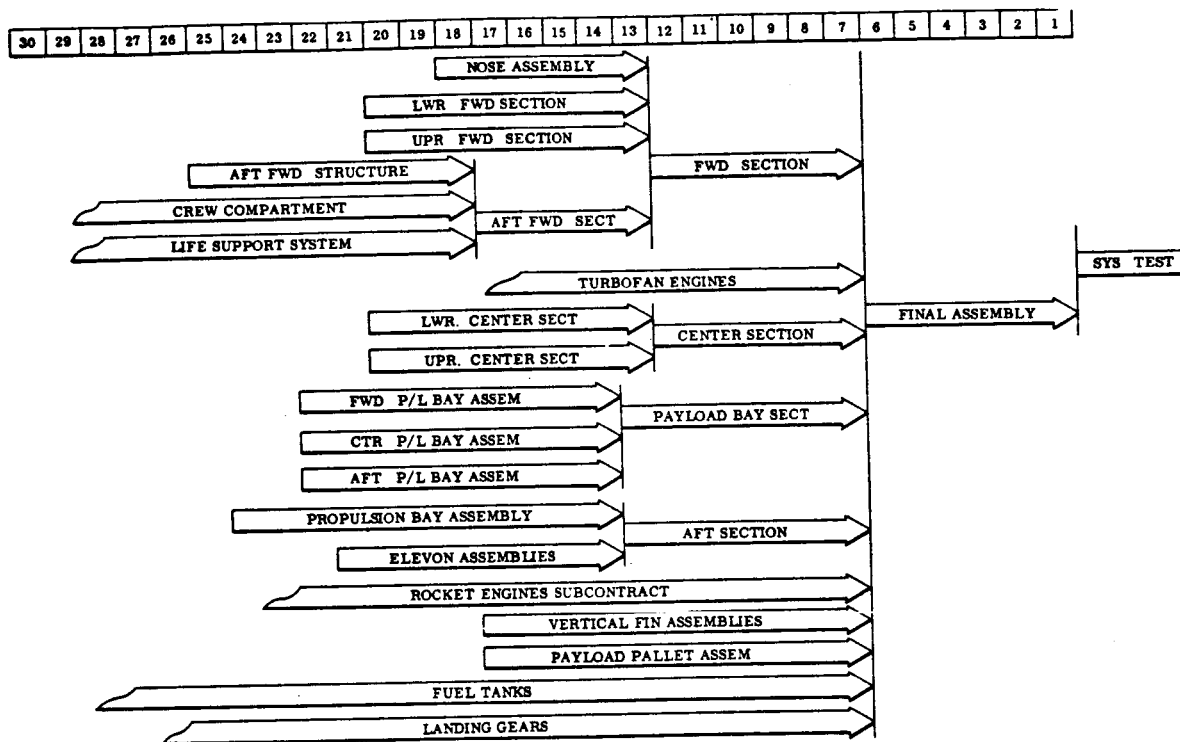


Fig. 2-18 Major Assembly Spans

## 2.9 COSTS, SCHEDULES, AND FACILITIES

The cost data are derived from a parametric cost analysis, in which the primary objective was to compare various design approaches in order to arrive at a minimum cost baseline configuration. The cost model in current use is based on a model originally developed by the Institute for Defense Analysis (IDA) for reusable space launch vehicles. This model was selected as a point of departure because of its good overall coverage and the compatibility of its required vehicle parameter input with the output of the MAGIC vehicle sizing program. However, the original IDA cost model tended to combine many cost elements and did not provide the definition of costs to the depth required and as defined in NASA "Specification for the Presentation of Cost and Schedule Plans for New Space Projects, Phase A (PPP)," dated January 30, 1969. Therefore, the original model has been substantially modified and expanded to provide an output that is more closely aligned to the work breakdown structure of the NASA specification.

RDT&E costs are summarized in Table 2-10. First unit cost of the flight system is \$137.7 million.

Table 2-10

### RDT&E COSTS (in millions of dollars)

	<u>Orbiter</u>	<u>Booster</u>	Ground Support Equipment	395
Structures	1536	1553	Launch Operations	5
Propulsion	637	19	Propellants	8
Other Subsystems	305	54	Launch Facilities	220
Facilities	208	208	Refurbishment	11
Test Hardware	282	435		
Total	<u>2968</u>	<u>2269</u>		
Grand Total . . . . .				<u><u>5876</u></u>

Average recurring costs per flight are presented in Fig. 2-19. These costs include amortizing the cost of the production flight vehicles over the total number of flights in the operational program. Figure 2-20 shows the distribution of recurring costs as a function of the number of operational flights in the program. The steps in both curves indicate points where higher quantities of production vehicles are required to support the increased launch rates of the larger flight programs.

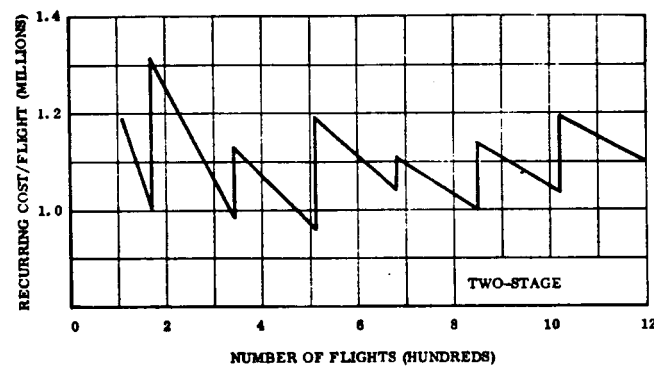


Fig. 2-19 Recurring Cost per Flight

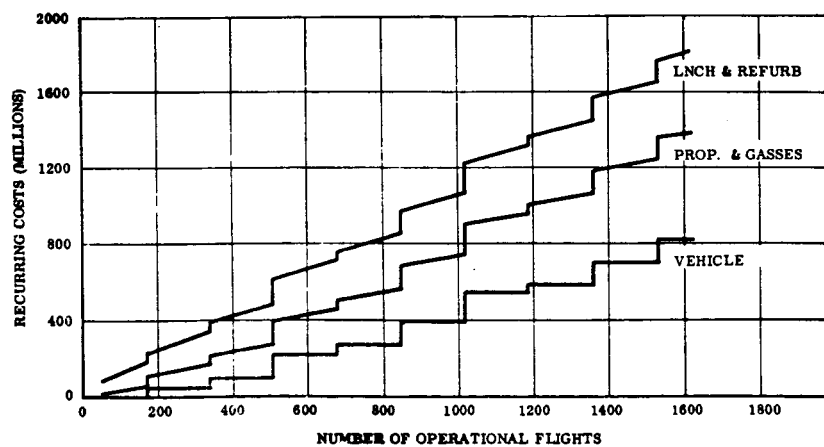


Fig. 2-20 Total Recurring Cost Breakdown

Requirements that influence design, manufacture, test, and operations have been identified; and preliminary schedules have been developed on the basis of these findings.

Key technical issues evaluated are as follows:

- Wing tunnel testing of candidate configuration concepts
- Thermal protection
- Recovery
- Flight control
- Integrated avionics design
- Hot flow testing of full-scale integrated structural components
- Reusable nose cap development
- Integrated onboard checkout

Schedules are based on IOC in 1976. The summary schedule is shown in Fig. 2-21.

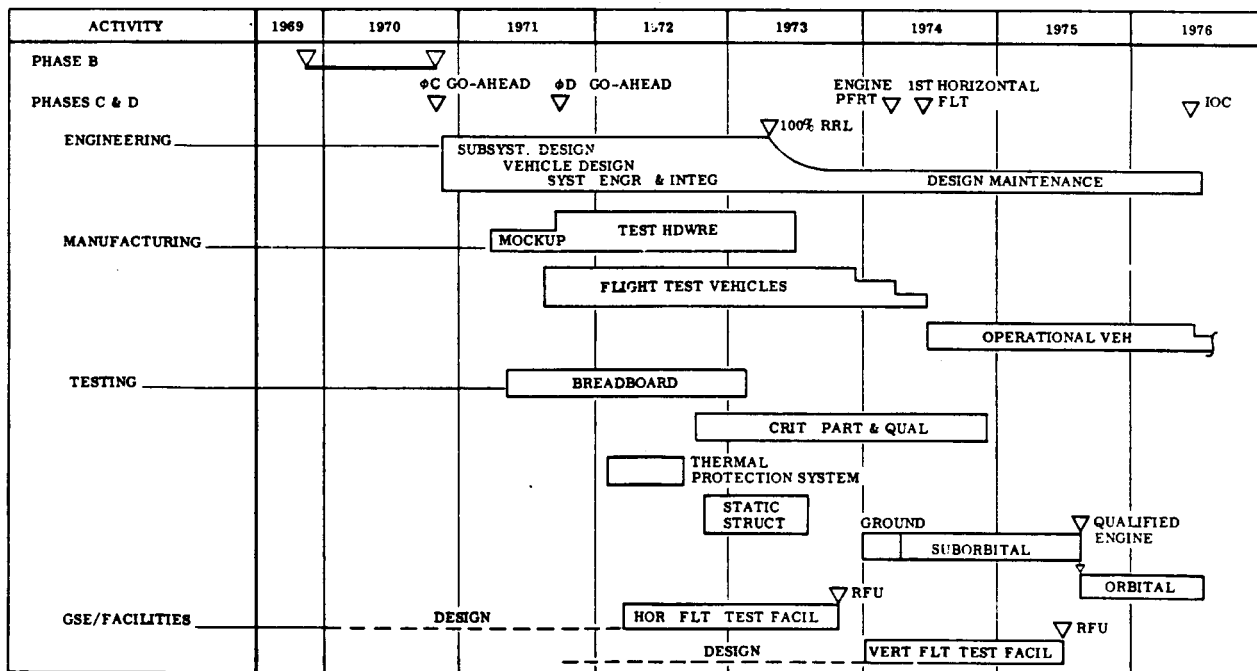


Fig. 2-21 Development Schedule

Facility requirements are expected to be well within the realm of existing designs.

It is planned that major subassemblies will be fabricated at the contractor's plant, that stage final assembly will take place at the suborbital flight test site, and that orbital flight testing of the complete vehicle will be conducted at the operational launch site.

Except for factory tooling, no new major facilities are required for manufacture. The Saturn I Test Facility at Michoud can probably be used for engine cluster testing after modification to accommodate the Space Shuttle engine. Suborbital testing can take place at a conventional facility, such as Edwards Air Force Base. Only the operational launch site entails new kinds of facilities, and they are for the most part adaptations of existing designs.